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AS-503A SPACECRAFT REFERENCE TRAJECTORY
VOLUME 1

By Donald B. Grammer, Roger H. Sanders,
and Walter Scott, Jr., Mission Analysis Branch,
and Kenneth A. Young, Rendezvous Analysis Branch

MISSION PLANNING AND ANALYSIS DIVISION
MANNED SPACECRAFT CENTER
HOUSTON, TEXAS

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PROJECT APOLLO

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AS-503A SPACECRAFT REFERENCE TRAJECTORY

By Donald B. Grammer, Roger H. Sanders,
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1.0 SUMMARY AND INTRODUCTION

1.1 Purpose

The spacecraft reference trajectory described in this document is for mission AS-503A and is designed to employ a manned-configured spacecraft. The mission profile has been established in an effort to satisfy the spacecraft mission objectives (ref. 1, 2, and 3) without violating any of the known launch vehicle or spacecraft constraints (ref. 4). The purpose of this document is to describe the complete mission profile that incorporates the revisions and refinements which have been officially identified since the publication of the preliminary reference trajectory (ref. 5).

1.2 Scope

This document is presented in three volumes and includes the following information:

a. Trajectory information, such as position, velocity, attitudes, lighting, and performance characteristics, in sufficient detail to permit definition of expected spacecraft subsystem performance.

b. Tracking and communications information to define the operational ground support equipment requirements.

<u>1.2.1 Volume I</u>.- Volume I contains a detailed description of the mission profile along with plotted and tabulated supplementary data arranged according to periods of activity.

<u>1.2.2 Volume II</u>.- Volume II contains the Apollo Reference Mission Program (ARMP) computer output and a key defining the output parameters.

<u>1.2.3 Volume III</u>.- Volume III contains radar acquisition and termination data and shadow time lines.

1.3 Acknowledgements

The following people made significant contributions to this document: Jack E. Hills, William Lamey, Burl G. Kirkland, and Robert F. Wiley of Mission Analysis Branch; Catherine T. Osgood of Rendezvous Analysis Branch; and Frank Evans of TRW Systems.

1.4 Mission Profile Summary

For the nominal trajectory simulation, launch is assumed to occur at 2100 Greenwich mean time (G.m.t.), September 30, 1967, from launch complex 39A on a flight azimuth of 72° from true north.

The mission profile has been divided into five periods of activities. These periods and the corresponding significant mission events are as follows:

o <u>First Period of Activities</u> (00:00:00 - 71:37:14 ground elapsed time (g.e.t.)

1. Launch
2. Insertion into earth-parking orbit
3. Platform alignment
4. Simulated translunar injection
5. Transposition, docking, and separation (SM RCS)
6. Platform alignment
7. Deep-space navigation
8. Simulated midcourse correction burn - first SPS burn
9. Deep-space navigation
10. Platform alignment
11. Simulated lunar orbit insertion - second SPS burn
12. Simulated translunar coast

- <u>Second Period of Activities</u> (71:37:14 - 120:30:00 g.e.t.)

13. First docked DPS Hohmann burn
14. Second docked DPS burn
15. Simulated lunar orbit coast (DPS power)

- <u>Third Period of Activities</u> (120:30:00 - 137:04:59 g.e.t.)

16. Lunar module separation (LM RCS)
17. Fire-in-the-hole (LM APS)
18. Concentric sequence initiation (LM APS)
19. Constant differential height maneuver (LM RCS)
20. Terminal phase initiation (LM RCS)
21. Terminal braking (LM RCS)
22. Capture and hard docking (LM RCS)
23. CSM separation (SM RCS)
24. APS burn to depletion

- <u>Fourth Period of Activities</u> (137:04:59 - 238:39:24 g.e.t.)

25. CSM phasing maneuver - third SPS burn
26. CSM constant differential height maneuver - fourth SPS burn
27. CSM terminal phase initiation (SM RCS)
28. Flyby
29. Simulated transearth injection burn - fifth SPS burn
30. Simulated transearth coast

- <u>Fifth Period of Activities</u> (238:39:24 - 239:21:07 g.e.t.)

31. SPS deorbit burn - sixth SPS burn
32. CM/SM separation
33. Atmospheric reentry
34. Landing

1.5 Mission Profile Evaluation

It is understood at the time of publication of this document that a reversion to four high elliptical orbits is required to alleviate the crew activity time line. Trajectory planning for four high elliptical orbits and two injection opportunities is proceeding for the operational trajectory.

The primary purposes of mission AS-503A are to demonstrate the capabilities of the launch vehicle, spacecrafts, crew, and ground support facilities to perform the lunar orbital rendezvous (LOR) mission operations while following a lunar landing mission (LLM) time line as closely as possible in earth orbit. The mission is designed for a 9-day duration.

In general this AS-503A spacecraft reference trajectory satisfies the mission objectives and does present as nearly complete a simulation of the LOR mission events time line as is possible in an earth-orbital environment. In the following instances, deviations from the LLM were made to comply with the AS-503A constraints, to enhance crew safety, or to provide more meaningful systems tests:

a. The second S-IVB burn, simulating translunar injection (TLI) is much shorter in order to limit the apogee altitude and, consequently, the service propulsion system SPS propellant required for circularization. This satisfies radiation, navigation, and Apollo guidance computer (AGC) scaling constraints and, at the same time, conserves the SPS propellant required for other maneuvers.

b. The transposition and docking attitude is a compromise that was made between the Block I spacecraft communications, the S-IVB communications, and the soft-docking lighting constraints.

c. The 472-second SPS circulatization burn is larger than the lunar orbit insertion (LOI) burn because of the 3950-n. mi. apogee required for the deep-space navigation exercise. It does, however, employ the same guidance as the LOI burn for the LLM.

d. The time difference between the lunar module (LM) descent propulsion system (DPS) burns and the first ascent propulsion system (APS) burn is about twice that of the LLM. The LM APS active rendezvous to simulate the ascent and rendezvous phases is similar in sequence, but also takes about twice as long to complete as rendezvous for the LLM.

e. The docked vehicle configuration became necessary when the requirements for full LM/DPS/APS duty cycles and continuous LM rescue capability by the command and service modules (CSM) were established. As a consequence of the docked vehicle configuration, the ΔV magnitudes of the DPS burns were about half of those for the LLM even though the burn times and propellant usage were quite similar.

f. Crew safety considerations dictated that the long duration APS burn be performed unmanned.

g. The transearth injection (TEI) burn follows about the same time line as the LLM, but is not of the same magnitude because of the SPS usage required prior to and following this burn.

h. The CSM active rendezvous with the spent DPS has two corrective SPS maneuvers which do not completely satisfy the ground tracking constraints. The geographic location of these SPS burns can be easily adjusted to obtain better ground coverage.

i. A shorter rendezvous plan together with a shorter crew operations time line for preparing the APS for the unmanned propellant depletion burn is being considered for the operational trajectory to improve the APS electrical power shortage in this reference trajectory.

2.0 SYMBOLS

AGC	Apollo guidance computer
AOT	alignment optical telescope
APS	ascent propulsion system
AS	Apollo Saturn
CDH	constant differential height

CM	command module
CSI	concentric sequence initiation
CSM	command and service modules
DPS	descent propulsion system
ECS	environmental control system
EPS	electrical power system
EST	Eastern standard time
ETR	Eastern Test Range
FITH	Fire-in-the-hole
g.e.t.	ground elapsed time
G.m.t.	Greenwich mean time
IU	instrument unit
IMU	inertial measurement unit
IP	impact point
LET	launch escape tower
LLM	lunar landing mission
LM	lunar module
LOI	lunar orbit insertion
LOR	lunar orbital rendezvous
LOX	liquid oxygen
MSFN	Manned Space Flight Network
PGNCS	primary guidance and navigation control system
RCS	reaction control system
SLA	spacecraft LM adapter

SM	service module
SPS	service propulsion system
TEI	transearth injection
TPI	terminal phase initiation
VHF	very high frequency

3.0 MISSION PROFILE GUIDELINES

The pertinent trajectory-related mission guidelines or constraints which influenced the trajectory shaping and time line are presented in this section for the convenience of the reader. Reference 4 contains the complete list of all known launch vehicle and spacecraft systems and operational constraints which had to be met to satisfy the mission objectives.

a. No limits were assumed on the continuous SPS burn time for the Block II engine.

b. The star-earth landmark sightings will be accomplished at an altitude of 3000 n. mi. or greater and will have a sunlit landmark.

c. The CSM scaling limits the apogee for this mission to 5400 n. mi. and the inertial velocity to 32 000 fps.

d. At all times during manned LM operations, the CSM shall maintain the capability to rendezvous with the LM to provide such assistance as may be required and will maintain the capability to perform the SPS deorbit maneuver.

e. It is highly desirable that the predicted orbital decay lifetimes for the LM spent DPS and APS not exceed 3 months.

4.0 SUMMARY OF INPUT DATA

4.1 Launch Vehicle Trajectory and Performance

The launch vehicle description, trajectory, weights and performance, targeting parameters, and other pertinent information appear in reference 6.

4.2 Spacecraft Weight and Performance

Table I shows spacecraft engine performance. This information and spacecraft weight data were obtained from references 7, 8, 9, and 10.

4.3 Spacecraft Aerodynamic Data

The command module (CM) aerodynamic coefficients used for the entry trajectory simulation were taken from reference 10, and are shown in tables II and III. In generating the entry trajectory, the U. S. Standard 1962 Atmosphere and the Fischer Earth Model of 1960 were used.

4.4 Spacecraft Guidance

Spacecraft guidance equations and logic were obtained from reference 11. The guidance modes employed during the simulation of each spacecraft burn are shown in table IV.

4.5 Ground Stations and Ship Locations

Ground stations available for the support of this mission, their tracking equipment, and their locations were obtained from reference 12 and 13. These data are summarized in tables V and VI. Suggested locations for the tracking ship are also indicated in table V. Four ships are required in order to operate within the trajectory constraints presented in reference 4.

4.6 Coordinate Systems Definition

Reference 14 provides a definition of the approved Apollo coordinate systems. These coordinate systems were used in the programs which generated this reference trajectory. Figure 1 presents vehicle configurations and the three main coordinate systems.

5.0 NOMINAL MISSION DESCRIPTION

5.1 First Period of Activities

The first period of activities, including Events 1 through 12, is described below and shown in figure 2. These activities simulate the earth launch, earth-parking orbit, second S-IVB burn to injection, transposition and docking, midcourse correction, simulated lunar orbit insertion, and translunar coast.

5.1.1 Launch (Events 1 and 2).- For simulation purposes launch is assumed to occur at 4 p.m. Eastern standard time (E.s.t.), on September 30, 1967, from launch complex 39A of the Merritt Island Launch Area (MILA). A 2-hour launch window is available prior to sunset to allow for unscheduled holds during the launch countdown. The spacecraft will liftoff on a 72° launch azimuth and will be inserted into a 100-n. mi. circular parking orbit by the S-IVB.

The launch phase is typical for the Saturn V configuration (ref. 6). An 11-second vertical rise is followed by a programmed pitchover which terminates at S-IC cutoff 150.6 seconds g.e.t. Staging is completed, and full S-II thrust is attained 6.5 seconds later.

Launch vehicle guidance is initiated 0.5 seconds after the S-II has reached full thrust. The S-IC/S-II interstage is jettisoned 23.7 seconds after S-II ignition, and the launch escape tower (LET) is jettisoned 5 seconds later. The S-II cutoff and S-IVB ignition occur 524.192 seconds g.e.t.

The launch phase is completed at S-IVB cutoff 695.28 seconds g.e.t. at insertion into the earth-parking orbit.

5.1.2 Earth Parking Orbit.- The earth-parking orbit phase will include two complete orbits to allow time for inertial measurement unit (IMU) alignments, systems checkouts, and ground-based orbit determination. The duration of the earth-parking orbit phase is approximately 3 hours.

Immediately after orbital insertion the vehicle is oriented so that the spacecraft longitudinal axis is aligned with the local horizontal, and the crew are in a heads-down position. This relative attitude is maintained for an IMU alignment which is performed in darkness, and will require approximately 17 minutes to complete. S-IVB liquid oxygen (LOX) venting will occur periodically after this maneuver.

The second IMU alignment (Event 3) will require approximately 52 minutes (ref. 15). This maneuver includes coarse alignment, fine alignment, and orientation for the burn. The alignment will be made mostly in darkness. Immediately after this maneuver, the required inertial attitude will be established for the S-IVB burn into the high-apogee ellipse.

5.1.3 Simulated Translunar Injection (Event 4).- Prior to the second S-IVB burn, two events occur: Spacecraft platform alignment and S-IVB ullage. The purpose of the second S-IVB burn is to simulate translunar injection for the LLM. The S-IVB will utilize the guidance equations to be used on a lunar landing mission, although the form of the target will be slightly modified.

The S-IVB burn injects the vehicle into an elliptical trajectory which satisfies the deep-space objectives of this mission.

Ignition of the S-IVB occurs at 3 hours, 16 minutes, 22 seconds g.e.t. The burn lasts 2 minutes, 30.5 seconds, and the vehicle traverses a central angle of 11° during the burn. Ignition occurs at 27.3° N latitude and 61.1° W longitude. Burnout occurs at 23.3° N latitude and 50.4° W longitude, at an altitude of 111.7 n. mi. The ΔV for the burn is 4086.35 fps. This second S-IVB burn places the vehicle in an elliptical orbit with an inclination of 32.6°, a perigee altitude of 107 n. mi., a period of 2.84 hours, and an apogee altitude of 3950 n. mi. Tracking coverage for this burn is from Merritt Island, Grand Bahama, Bermuda, Antigua, and the insertion ship. The burn will have coverage from 17 minutes before to 3 minutes after its occurrence. Attitude data are shown in table VII. Trajectory parameters during the high apogee elliptic orbits are presented in figures 11 and 12.

5.1.4 Transposition, Docking, and Separation (Event 5).- The events of the first elliptical orbit begin at burnout of the S-IVB. LOX venting is planned to occur within 2 minutes after S-IVB cutoff as required for blowdown. Table VIII gives a detailed time line and figure 7 presents plotted data of the transposition and docking. Communication coverage is shown in figure 10. The spacecraft (S-IVB/IU/LM/CSM) will establish 0° attitude between the spacecraft +X axis (forward) and local horizontal and maintain an orbital pitch rate to remain along the local horizontal. Approximately 15 minutes after cutoff an inertial pitch maneuver of -103.5° and roll maneuver of -5.2° are required to move the S-IVB to the solar orientation of 30° with respect to S-IVB X axis for the desired illumination of the vehicles. This orientation takes approximately 6.5 minutes at the pitch rate of 0.3 deg/sec. This inertial attitude is maintained throughout the transposition and docking exercise. +X translation is initiated using 1 fps from the service module (SM) reaction control system (RCS) quads 22 minutes after cutoff. This maneuver terminates when the relative range reaches 100 ft and takes approximately 1 minute. Another 1 fps is used to null out the 1 fps introduced during withdrawal. After entering sunlight (26 minutes after S-IVB cutoff), the CSM is oriented for docking. This requires a pitch of 180° and a roll of -60°. The crew will have the sun above and behind them for docking. The next maneuver is the 1 fps transposition back to the LM.

Immediately after transposition back to the LM, soft docking (capture) will occur which requires approximately 10 minutes. Therefore, approximately 20 minutes is used in transposition and soft docking. The hard docking will require 45 minutes following the soft docking. A crew member will crawl into the docking tunnel, begin the hard-docking procedure of throwing eight latches, and crawl back into his couch.

The ΔV allotted for LM extraction from the S-IVB is 3 fps and is provided by the SM RCS. The LM is oriented forward in the direction of motion. After LM extraction, the S-IVB is oriented and vented (blowdown) to give maximum separation between the S-IVB and the CSM/LM. Blowdown of the S-IVB is nominally planned to occur 2 hours after S-IVB cutoff, but will be initiated immediately after LM separation with a ground command in the event of an early LM separation. The separation distance between the S-IVB and CSM/LM is nearly 19 n. mi. until the apogee of the second ellipse. After the midcourse correction maneuver (Event 8), the distance increases to approximately 150 n. mi. A 1° plane change is included in the simulated LOI burn (Event 11) reducing the probability of recontact with the S-IVB.

5.1.5 Platform Alignment (Event 6).- The second ellipse begins with a platform alignment in the earth's shadow. Alignment requires approximately 31 minutes.

5.1.6 Deep-Space Navigation (Event 7).- Star-landmark sightings will begin at 3000-n. mi. altitude and will require one, three-axis maneuver to start the sightings and a pitch and yaw to change target. Six different landmark sightings (two targets) are planned for this phase, each requiring 6 minutes. After the sightings and 1 minute before apogee an orientation will be performed to align the body axis for the first SPS burn.

5.1.7 Simulated Midcourse Correction Burn (Event 8).- At apogee of the second ellipse, an SPS burn will be performed to raise perigee to 150-n. mi. altitude and simulate the midcourse correction maneuver. A minimum perigee of 150 n. mi. is required to get the minimum required coverage for the SPS burn to simulate LOI (Event 11) while retaining the required RCS deorbit capability. This burn starts at 7:31:48 g.e.t. and continues for 8 seconds. The ΔV required for this in-plane burn is 55 fps, resulting in a perigee of 148-n. mi. altitude. The burn occurs at 57° E longitude and is tracked by Carnarvon (table IX(a) and fig. 8(a)). The guidance used for this burn is the external ΔV mode. Trajectory parameters during this phase are shown in figure 13.

5.1.8 Deep-Space Navigation (Event 9).- Immediately after the first SPS burn, another set of six star-landmark sightings will begin. A total of 36 minutes is allowed for these sightings. The same altitude requirement exists for these sightings as for the previous set. The sightings will end at 3000-n. mi. altitude. Star-horizon sightings could possibly be made for about 13 minutes; then, platform alignment will be initiated. It may also be possible to make star-horizon sightings in the first ellipse between the final separation from the S-IVB and the initiation of platform alignment.

5.1.9 Platform Alignment (Event 10).- At 8 hours, 20 minutes g.e.t., the platform alignment begins for the simulated LOI maneuver. The schedule allows 31 minutes for this maneuver. The platform alignment begins with a gross attitude maneuver to bring the selected stars into the sextant field of view. The RCS propellant budget, shown in table I(c) allows for one, three-axis maneuver. The fine alignment which follows requires not more than two, two-axis maneuvers. The fine alignment will bring the stars into the crosshairs of the sextant. Prior to the simulated LOI burn, the crew must switch from wide deadband attitude hold to fine deadband attitude hold to perform the burn. After platform alignment is completed, the spacecraft is oriented to the attitude required for the burn.

5.1.10 Simulated Lunar Orbit Insertion (Event 11).- The second SPS burn to lower apogee to 200 n. mi. begins at 8 hours, 52 minutes, 48 seconds g.e.t. The SPS engine burns for 7 minutes, 52 seconds, providing 4069 fps ΔV (Lambert's guidance). The characteristics of burnout are a flight-path angle of 0.14°, a period of 1.5 hours, an altitude of 148 n. mi., an inclination of 31.9°, and eccentricity of 0.007. This burn is an approximate simulation of the lunar orbit insertion. The radar tracking for this burn as shown in table IX(a) is from Hawaii and the injection ship. The operational requirement for 2 minutes of tracking before and after the burn is satisfied. For more detailed tracking information, refer to the bar chart in figure 9, which shows Manned Space Flight Network (MSFN) coverage, major events, and daylight-darkness time line for all periods of activities. Trajectory parameters during this phase are presented in figure 14.

The groundtrack for the first day (fig. 8(a) shows the S-IVB injection burn and the two perigee burns with associated tracking requirements. The figure demonstrates the capability of an open-ended mission in which an orbit is provided whereby the crew could circularize and continue on with the mission. This capability is desirable because of the uncertainty in the amount of radiation that may be encountered. Due to the variation of the flux fields at this altitude and the possible dispersion of the nominal trajectory, the open-ended type mission was selected as highly desirable.

The radiation dosage to the crew during the two elliptical orbits has been calculated to be under the operational limit (ref. 16 and 17). A measurement of the radiation dosage will be monitored during the mission. If this measurement indicates excessive radiation, the spacecraft can circularize at the next perigee passage.

5.2 Second Period of Activities

The LM DPS maneuvers, Events 13 and 14, and the simulated lunar orbit coast period, Event 15, are described below. Figure 3 shows a schematic of these events.

5.2.1 First Docked DPS (Hohmann) Burn (Event 13).- Event 13 simulates the Hohmann descent maneuver of the LLM as well as prepares for the LM main powered-descent simulation (Event 14).

A 6-second ullage burn immediately prior to ignition of the DPS engine consumes 8.6 lb of RCS propellant and is achieved by the use of four 100-lb thrust engines. The DPS burn itself is a near Hohmann type and lasts for 30 seconds. During this period, 243 lb of propellant are consumed, which represents 38 fps ΔV for docked vehicle configuration. The ignition of the burn occurs at 71:37:12 g.e.t. at an altitude of 189 n. mi., a latitude of 28° S, and a longitude of 150° E. The station tracking the burn is Canberra, and the entire maneuver is performed in sunlight.

The attitude orientation during the burn is retrograde along the local horizontal and in the plane of motion. The orbital characteristics resulting from the maneuver are a perigee altitude of 125 n. mi., an apogee altitude of 192 n. mi., and a period of 1.508 hours.

Following the first docked DPS burn, a coast period begins at apogee (192-n. mi. altitude) and terminates 2 hours, 5 minutes later (138-n. mi. altitude) when Event 14 is initiated. During this phase a wide deadband hold (5°) is maintained. Other activities during this phase include a gross and fine platform alignment, orientation, navigation, communication, and subsystem checkout to prepare for Event 14.

5.2.2 Second Docked DPS burn (Event 14).- Event 14 is a continuous, long duration (12.08 minutes) DPS burn (docked), most of which is applied out of plane. The docked vehicle configuration was chosen to eliminate the problem of requiring the CSM to have LM rescue capability at any time during vehicle separation. This requirement along with the requirement to obtain a full LM DPS duty cycle made the docked vehicle configuration necessary. External ΔV is the guidance mode used to control the burn. This guidance logic holds a constant inertial thrust attitude until a specified ΔV is achieved by the burn. The target ΔV vector for Event 14 was chosen so that the vehicle would end the burn with a small (approximately 50 fps) additional forward component of velocity. This additional forward component of velocity was a precautionary measure to insure that any dispersions during the burn would not place the spacecraft on a reentry trajectory. Trajectory parameters for this phase are shown in figure 15. Figure 15(e) shows the DPS attitude profile for this long duration burn. The angle β , shown in figure 15(e), is the yaw angle, or angle between the thrust vector and vehicle plane of motion. Figure 15(f) shows the thrust level profile for the long duration DPS burn. After the initial start phase, during which 30 percent thrust is applied for 3 seconds and 10 percent thrust for 23 additional seconds, the DPS engine is held at maximum thrust for 317 seconds, dropped to 60 percent for 215 seconds, and finally dropped to 30 percent where it is held for the remaining 167 seconds. This thrust profile simulates the LLM LM duty cycle as closely as feasible in earth orbit.

Event 14 is shown schematically and by a groundtrack in figures 3 and 8(b), respectively, and is tabulated in table IX(b). The maneuver begins at 73 hours, 45 minutes g.e.t. at an altitude of 138 n. mi., a latitude of 32° N, and a longitude of 100° W. The maneuver ends 12.08 minutes later at an altitude of 127 n. mi., a latitude of 25° N, and a longitude of 48° W. Stations tracking are Texas, Cape Kennedy, Bermuda, and the insertion ship. The maneuver begins in sunlight and ends in darkness.

Immediately before the initiation of the long DPS burn, a 6-second ullage burn is performed by the RCS engines and consumes 8.6 lb of propellant. The DPS propellant consumed by Event 14 is 16 852 lb, leaving

260 lb (1.5%) in the DPS tanks. The 16 852 lb of propellant used represents 3148 fps ΔV with the docked vehicle configuration. The resulting orbital characteristics are a perigee altitude of 127 n. mi., an apogee altitude of 224 n. mi., and a period of 1.517 hours.

5.2.3 Simulated Lunar Orbit Coast (Event 15).- This phase begins after Event 14 and ends approximately 46.5 hours later when the third period of activities is initiated. The spacecraft remains in a docked configuration and in the same orbit achieved by Event 14. During this coast period, a wide deadband hold is maintained by the SM RCS. Trajectory related activities during this period include navigation, communication, and a platform alignment in preparation for Event 16.

5.3 Third Period of Activities

5.3.1 Lunar Module Separation (Event 16).- At 120:30:00 g.e.t. the two-man LM separates from the CSM with a small retrograde LM RCS maneuver of about 1 fps while over ship 1 (Mercury). This thrust will place the LM about 600 ft below and 400 ft behind the CSM at the time of the fire-in-the-hole (FITH). A diagram showing the events during the third period of activities is presented in figure 4.

5.3.2 Fire-In-The-Hole (FITH) (Event 17).- At 120:42:25 g.e.t. the LM will begin a 22-second APS burn (FITH) to partially simulate the ascent from lunar surface. The magnitude of this maneuver was selected to produce the necessary phasing between the LM and the CSM to perform a CSM-active rendezvous and to provide a range adequate for the rendezvous radar test. The DPS is jettisoned at the initiation of the FITH and will have a nearly impulsive velocity of about 17 fps imparted to it. The details of this impulse and the subsequent CSM flyby of the DPS are discussed in section 5.4. Figure 24 shows the relative trajectory between the LM and DPS from FITH to the APS burn to fuel depletion.

The FITH maneuver is done 9 minutes before sunset while over Florida; MSFN coverage is possible from Merritt Island Launch Area, Texas, and Grand Bahama Island. Figure 8(c) shows the groundtracks during this phase. The LM will be pitched down 62.3° (fig. 16) to begin the 225.9-fps burn; thus, the resultant incremental velocity will be 102.6 fps in the direction of motion and 201.2 fps down. The perigee of the LM orbit will be essentially unchanged in inertial position, thereby simplifying the subsequent concentric sequence initiation (CSI) and constant differential height (CDH) maneuvers. The LM apogee is raised to 291 n. mi., and perigee is lowered to 117 n. mi., which is about 11 n. mi. lower than the CSM perigee. The purpose of the large change in the LM orbit is to produce an adequate range between the LM and CSM for the LM rendezvous radar test. As seen in figure 17, the range reaches a maximum of 298 n. mi. and is greater than 260 n. mi. for about 30 minutes prior to the CSI maneuver, thus meeting the requirements for the rendezvous radar tests. After FITH the LM will move ahead of the CSM for about 41 minutes, but due to its higher apogee the LM will eventually go above and drop behind the CSM (fig. 21 and 23). The look angle from the LM to both the CSM and the DPS during the period from FITH to CDH is approximately the same (fig. 19); thus, if the LM to CSM radar does not provide very accurate pointing data, it may be problem for the LM Apollo optical tracker (AOT) to distinguish between the CSM and DPS especially at times near sunrise and sunset when both vehicles are illuminated by reflected sunlight. In darkness, of course, the DPS will not be visible since it will not have lights. Figure 20 indicates the sun angle off the line of sight from the LM to the CSM and from the CSM to the LM during the rendezvous.

5.3.3 Concentric Sequence.- In order to provide a large range (298 n. mi.) between the LM and CSM and yet retain a desirably low differential altitude (ΔH) of between 10 and 15 n. mi., it was necessary to schedule the FITH, CSI, and CDH maneuvers each a full orbit apart to allow for adequate LM separation and catch up. Therefore, the CDH was performed at the second apsis crossing point. Although this option is in the CSI logic, the nominal lunar orbit rendezvous plan does not currently make use of it.

5.3.3.1 Concentric Sequence Initiation (Event 18): At 122:26:32 g.e.t. the LM will initiate the CSI burn (computed onboard) of 164.4 fps retrograde. This horizontal maneuver provides the proper phasing between the LM and the CSM (fig. 18) for the CDH maneuver. The CSI burn is done at LM perigee to preserve the differential altitude of about 11.5 n. mi. The APS will be burned for about 15 seconds while the LM is being tracked by Antigua and the insertion ship (Vanguard). The resultant LM orbit has a 195-n. mi. apogee and a 116-n. mi. perigee. At the time of CSI the LM trails the CSM by about 278 n. mi.

5.3.3.2 Constant Differential Height maneuver (Event 19): The CDH is done at perigee at 123:55:57 g.e.t. to produce a LM orbit about 11.5 n. mi. below the CSM. Due to the elliptical orbits, the ΔH will vary from about 11.3 n. mi. to 12.5 n. mi. The 32-fps posigrade burn (26 seconds) is done with the RCS as the LM passes over Florida and is tracked by the Merritt Island Launch Area and Grand Bahama Island. This maneuver raises apogee to about 206 n. mi. and occurs as the vehicles enter darkness.

5.3.3.3 Terminal Phase Initiation (TPI) (Event 20): Since the current desired lighting for TPI occurs about 11 minutes prior to sunset with a pitched up attitude of 26.7°, (relative elevation angle of CSM measured with respect to the LM) the LM must travel about 74 minutes after CDH to arrive at this condition. At the time of TPI, 125:10:19 g.e.t, the velocity increment needed is only about 24 fps to place the LM on an orbit to intercept the CSM after about 140° of CSM orbital travel. Therefore, the LM RCS is burned for 20 seconds at a pitch up of about 33.1° from the local horizontal. Figure 23 shows the relative motion during terminal phase. Tracking coverage of the TPI will be from Hawaii, although the radar elevation angle at the start of the burn may be below 5°.

5.3.3.4 Terminal Braking (Event 21): The manual LM braking schedule followed is similar to the now-eliminated automatic gates shown below:

Range	Maximum allowable range rate
5 n. mi.	80 fps
3 n. mi.	30 fps
1 n. mi.	10 fps
500 ft	5 fps

Since the closing rate for an intercept from a ΔH of about 11.5 n. mi. is about 24 fps, the first two gates are passed. At a range of 1 n. mi. 125:42:00 g.e.t., the LM is approximately 0.70 n. mi. below and 0.71 n. mi. ahead of the CSM. At this time the RCS is used to apply a posigrade ΔV of 13 fps, thus reducing the range rate to 10 fps. The thrust angle is about 50° below the local horizon. After closing to 500 ft, the LM RCS applies 8 fps posigrade ΔV with a thrust angle of 29° below the local horizontal. Station keeping is considered to begin about 2.5 minutes later at 125:53:00 g.e.t. as the vehicles pass south of Ascension. Soft docking is initiated about 13 minutes later.

5.3.4 Hard Docking (Event 22).- By the time the vehicles pass Guam at 126:35:00 g.e.t., the LM should be hard docked. The CSM-LM perigee and apogee will be approximately 128 and 217 n. mi., respectively, until the CSM separation and APS burn to depletion, which occurs about 1.5 hours after hard docking.

5.3.5 CSM Separation (Event 23).- At 128:00:00 g.e.t., the CSM will separate from the unmanned LM in preparation for the APS burn to fuel depletion. The SM RCS is used to apply a 1-fps maneuver to separate the CSM from the LM out of plane to the south such that the vehicles are about 1000 ft apart at the time of the APS burn.

5.3.6 APS Burn to Depletion (Event 24).- At 128:25:14 g.e.t. over Hawaii and just prior to sunset, the APS begins a burn of 5552 fps out of plane to the north. The APS is yawed left 95.7° so that at the end of the 365-second thrust, the resultant velocity increments are 30.4 fps in the direction of motion and 5551.9 fps out of plane. The inclination of the APS orbit is lowered to 22.5°, and perigee and apogee become 128 and 234 n. mi., respectively. The resultant separation at the end of the burn is about 149 n. mi. with the LM to the northeast of the CSM. Figure 25 illustrates the separation between the CSM and the APS after the APS burn, and shows that there is no possiblility of recontact.

5.4 Fourth Period of Activities

The mission requirement to perform a CSM-active rendezvous from above can be satisfied only by a catch up and close flyby of the DPS during the fourth period of activities since the LM APS is to be burned to depletion, causing a large separation and even the possibility of APS disintegration.

At the time of the FITH maneuver, 120:42:25 g.e.t., the DPS is jettisoned at about 17 fps (fig. 5). Due to the LM attitude for the FITH, the velocities are imparted opposite the direction of motion and upward, about 7 fps and 15 fps, respectively, thereby causing the DPS to fall behind and to move above the CSM and LM initially (fig. 22). After about 36 minutes, however, the DPS moves below and ahead of the CSM at the rate of about 26 n. mi. per orbit. Thus, after about 11 orbits (at the time of the CSM phasing maneuver) the DPS will lead the CSM by 287 n. mi. (fig. 26).

5.4.1 CSM Phasing Maneuver (Event 25).- At 137:04:59 g.e.t. the CSM will perform an SPS maneuver to begin closing on the DPS which leads by 287 n. mi. The ground-computed maneuver will produce the desired height (CSM 5 n. mi. above) and phase offset (CSM 57 n. mi. ahead) at the time of the CSM CDH maneuver about six orbits later. It is done eight orbits after the APS burn to depletion to provide ample time for ground-radar skin tracking of the DPS and to allow one sleep period on the CSM. The maneuver occurs over the Atlantic with possible low coverage by Antigua (fig. 8(d)). The resultant CSM perigee and apogee after the 59.4-fps burn (2.6 sec) are about 109 and 219 n. mi., respectively. The relative motion between the CSM and the DPS can be examined in figures 30(a) and 30(b).

5.4.2 CSM Constant Differential Height (Event 26).- The SPS burn of 77.35 fps will be initiated at 145:45:58 g. e. t. as the CSM passes over Carnarvon. The resultant CSM 133-by 215-n. mi. orbit will be about 5 n. mi. above that of the DPS until TPI, 92 minutes later. The crew will take sextant sightings on the DPS when possible in order to determine the DPS orbit and thus calculate TPI. Since the DPS will not have any lights, these sightings will be possible only during the daylight period from about 145:40:00 to 146:35:00 g.e.t. as the range decreases from 60 to about 32 n. mi. Figures 28(a) and (b) and 29 show the look angles to the DPS and to the sun, respectively. The groundtrack is shown in figure 8(e).

5.4.3 CSM Terminal Phase Initiation (TPI) (Event 27).- At 147:18:27 g.e.t. the CSM performs the RCS-TPI burn of 10.3 fps (pitched down 27.7°) to place the CSM on an orbit to intercept the DPS after about 140° of LM orbital travel (fig. 27). The maneuver occurs just prior to the Carnarvon pass about 7 minutes after sunrise. This lighting condition was chosed to insure a daylight flyby of the unlighted DPS about 35 to 40 minutes later.

5.4.4 Flyby (Event 28).- The CSM will approach within 0.3 n. mi. of the DPS at about 140:00:00 g.e.t. (fig. 31). The closest approach occurs as the vehicles pass over Guaymas. The decision was to forego an actual rendezvous (braking) since the SM RCS propellant required would probably exceed the margin budgeted for deorbit backup requirements. Thus, the CSM will continue past and move away from the DPS at about 23 n. mi. per orbit (fig. 30(b)).

5.4.5 Simulated Transearth Injection (TEI) burn (Event 29).- The fifth SPS burn, at approximately 149 hours, 10 minutes g.e.t., is a simulation of the transearth burn of the AS-504 mission. The burn follows the same sequence, although the ΔV is about one-third that of the AS-504 TEI burn. The ΔV used for this burn is 709 fps and leaves a sufficient propellant margin for the deorbit burn. This burn has three major functions. One function is to simulate the AS-504 time line. The second is to change the basic orbit for better SM RCS deorbit capability. The third function is to move the line of apsides to get better lighting at landing in the western Atlantic.

All the radar tracking of the simulated TEI burn is done over Guam. This burn occurs in daylight and terminates at an altitude of 193.7 n. mi, a latitude of 15.86° S, and longitude of 138° W. The ullage propellant required for this burn is approximately 15 lb using two jets. The SPS burn duration is approximately 29 seconds and consumes 1855 lb of propellant. Trajectory parameters for this phase are shown in figure 32.

5.4.6 Simulated Transearth Coast (Event 30).- The 89-hour coast period simulates the AS-504 transearth coast period. No major activities are planned for this period. Other expermiments are required on this mission and can be done during this activity, but will require a better definition of the objectives for a simulation. However, it should be noted that at this point, only 816 lb of SM RCS propellant remain for use. Of this amount, 751 lb must be reserved for an RCS backup deorbit contingency.

5.5 Fifth Period of Activities

5.5.1 SPS Deorbit Burn (Event 31).- At the end of the 89-hour simulated transearth coast at 238:39:24 g.e.t., the CSM is oriented to the SPS deorbit attitude. The orientation (fig. 33) is such that the CSM is in a head-ups, retrograde attitude and pitched downward 49° with respect to the local horizontal. Such an orientation allows the crewmen to observe the horizon during the deorbit burn. The ΔV imparted to the spacecraft is 370 fps and the SPS propellant consumed is 915 lb The 14-second burn, initiated at 238:41:24 g.e.t., places the CSM on an earth-intersecting ellipse and produces an inertial flight-path angle of -1.7° and an inertial velocity of 25 750 fps at the entry interface altitude of 400 000 ft. The deorbit burn is designed so that splashdown occurs near the center of the western Atlantic primary recovery area. Radar coverage for the burn is furnished by the entry ship, Watertown, located at 10° N latitude and 180° longitude. The mission events and groundtracks during the fifth period of activities are shown in figures 6 and 8(f), respectively.

5.5.2 CM/SM Separation (Event 32).- A coast of 22.7 minutes follows the deorbit maneuver until the entry interface altitude is reached. Figure 34 shows trajectory parameters during this coast period. Approximately 5 minutes prior to reaching this altitude, the SM is jettisoned and the CM maneuvers to the desired entry orientation. The SM is jettisoned at a pitch attitude of approximately 60° above the local horizontal by burning the -X SM RCS jets to propellant depletion. Following SM jettison, the CM is oriented to the desired entry attitude while coasting to the entry interface.

5.5.3 Atmospheric Reentry (Event 33).- The CM reenters 9 days, 23 hours, 4 minutes and 6.5 seconds after lift-off. The reentry point is at an altitude of 400 000 ft with a geodetic latitude and longitude of 28.00° N and 89.43° W, respectively. Landing occurs about 17 minutes later, approximately 800 n. mi. southeast of Bermuda at a geodetic latitude of 21.3° N and a longitude of 58.60° W. The local time at landing is 5:21 p.m. The nominal reentry groundtrack and maneuver envelope are illustrated in figure 35. The reentry range is approximately 1730 n. mi., and the reentry time from 400 000 ft to drogue chute deployment is 10.7 minutes. There is an additional 6.3 minutes from drogue chute deployment to splashdown. Table X presents the reentry state vector.

5.5.3.1 Reentry Targeting: Figure 36 presents the overshoot and undershoot boundaries for the AS-503 mission. The reentry corridor is based on a reentry reference altitude of 400 000 ft. Line A in figure 36 defines the set of velocities and corresponding flight-path angles that result in a reentry trajectory which reaches a peak altitude of 400 000 ft after the initial reentry into the atmosphere. This line is based upon a lift-vector-down spacecraft orientation until a load factor of 0.2 g is reached, followed by a zero-lift orientation (90° bank angle) until peak altitude is reached. This skip-out boundary is a conservative line based on a lift-to-drag ratio (L/D) of 0.40 and a 60° winter, thin atmosphere as defined in reference 18. Line B defines an overshoot boundary based on a constant flight time from 400 000-ft altitude to splashdown of 40 minutes of which 7 minutes is allowed for parachute descent. The spacecraft is held in a lift-vector-up attitude throughout reentry. The 40-minute reentry time defines the environmental control system (ECS) limitation which is a function of CM battery lifetime. The battery capability of the CM is based on a 45-minute flight time after electrical power system (EPS) switchover from the SM to the CM power source with 48 hours allowed for post landing time. The 5-minute difference is the time from CSM separation to reentry interface. The two undershoot boundaries, lines C and D are the combination of reentry velocities and flight-path angles that result in load factors not greater than 10 g during reentry based on zero-lift trajectory profiles. Line C is for a L/D of 0.4, and line D is for a reduced L/D of 0.3. Both were generated using a 60° winter, thin atmosphere. The target line E in figure 36 represents a line which is approximately centered in the reentry corridor. Also shown in figure 36 as dashed lines is a conservative estimate of the reentry flight-path angle dispersions based upon a manual deorbit mode as discussed in reference 19. Even with these dispersions a safe reentry is possible.

5.5.3.2 Spacecraft Data: The trim aerodynamic coefficients for the spacecraft are shown in table II. Above Mach 6 the CM has a constant lift-to-drag ratio, L/D, of 0.353 and a ballistic coefficient, W/C_DA, of 76.03 lb/ft^2 for the 12 000-lb command module. The CM center of gravity and aerodynamic characteristics as a function of angle-of-attack and Mach number were obtained from reference 10 and were used in computing the trim aerodynamic coefficients. The trim aerodynamic data correspond to a center-of-gravity position of 1040.2 inches in X, 0.0 inches in Y, and 5.8 inches in Z, where the X, Y, Z coordinates are defined in figure 37. The reference area used in these computations is 129.35 ft^2. Figure 37 also shows the aerodynamic force geometry of the spacecraft. The lift coefficients, C_L, and drag coefficients, C_D, are shown in the positive direction. The aerodynamic coefficients shown in table II do not include coefficient variations due to center-of-gravity caused by attitude control propellant consumption and heat shield ablations.

Parachute drag data for the drogue and main chutes are presented in table III. These data were obtained from reference 10 and are based on drogue chute deployment at 23 500-ft altitude and main chute deployment at 10 200-ft altitude. The forward heat shield, ablator burnoff, and about one-third of the attitude control propellant are assumed jettisoned at drogue chute deployment, thus reducing the spacecraft weight to 11 364 lb. The drogue chute and remaining attitude control propellant are assumed jettisoned at main chute deployment which again lowers the spacecraft weight to 11 127 lb.

5.5.3.3 Reentry Simulations: The reentry trajectory simulation was made with a four degrees-of-freedom computer program. Three degrees-of-freedom define the translational motion of the spacecraft center of gravity, and the fourth degree-of-freedom defines the spacecraft roll attitude. The reentry program includes a simulation of the roll channel of the stabilization and control system (SCS) and assumes the spacecraft to be trimmed in pitch and yaw. In conjunction with this, the reentry guidance logic defined in reference 11 was used to simulate the guided reentry trajectory.

The aerodynamic heating during reentry is an arithmetic sum of the convective and radiative heating. The convective heating rate is defined in reference 20. The radiative heating rate, was determined by using a table look-up routine which was developed by the Aerodynamic Branch of the Advanced Spacecraft Technology Division. A multiplying factor of 0.5 was used to make the radiative heating rate compatible with the point on the heat shield for which the convective heat was determined.

5.5.3.4 Reentry: The nominal reentry maneuver envelope and ground-track and significant events are shown in figure 35(a). The maneuver envelope was determined by holding the CM lift vector at a series of constant bank attitudes with the bank angle varied between 0° and ±90°. Figure 35(b) presents the SM maneuver envelope.

Figure 38 presents the altitude-range profile of the nominal reentry trajectory. The primary trajectory and guidance events are indicated on this figure. Figure 39(a) depicts the VHF, S-band, and C-band communications blackout regions of the reentry trajectory. The data for the blackout regions was obtained from reference 21. Figure 39(b) shows the radar tracking stations which may be used during reentry. During the blackout period the stations must have skin-track capabilities. This figure shows which stations will be skin-tracking the spacecraft.

Figure 40 presents significant reentry trajectory parameters as a function of time from lift-off. The maximum total heat rate obtained was 56.3 Btu/ft^2/sec with a total heat load of 13 571 Btu/ft^2. The maximum load factor experienced for this trajectory was 2.96 g. Figure 40(d) shows the nominal AGC display key (DSKY) quantities with the display beginning at 0.2 g during a nominal reentry based on this trajectory. Figures 40(e) and 40(f) show significant parameters for the drogue and main chutes based on a normal deployment sequence.

5.5.4. Landing (Event 34).- The mission is terminated at 239:21:07 g.e.t. with CM splashdown at 21.3° N latitude and 58.6° W longitude. The total ground range from entry interface to landing is approximately 1730 n. mi.

The estimated SM ballistic impact point (IP) coordinates are 25.3° N latitude and 73.7° W longitude. The approximate ground range of the SM impact point is 150 n. mi. from the nearest land mass (Bahama Islands) and 930 n. mi. from the CM landing point.

TABLE I.- SPACECRAFT ΔV BUDGET

(a) CSM/SPS ΔV Budget

Initiation, hr:min:sec g.e.t.	Event	Vehicle configuration	ΔV, fps	Fuel used, lb	Duration, min:sec
7:31:32.325	First SPS burn at apogee to raise perigee to 150 n. mi.	CSM/LM	54.7	491	7.733
8:52:48.293	Second SPS burn to simulate lunar orbit insertion	CSM/LM	4069.79	29 995	7:50:26
137:04:59	Third SPS burn to get phasing with descent stage	CSM	59.58	164	2.578
145:45:58	Fourth SPS burn to get phasing with descent state	CSM	77.52	212	3.32
149:10:24	Fifth SPS burn to simulate transearth injection	CSM	709.26	1856	29.218
238:41:24.4	Deorbit burn	CSM	370.056	913	14.413
Total fuel consumed, lb				33 631	
Reserve, lb				1442	

TABLE I.- SPACECRAFT ΔV BUDGET - Continued

(b) LM DPS ΔV Budget

Initiation, hr:min:sec g.e.t.	Event	Vehicle configuration	ΔV, fps	Fuel used, lb	Duration, min:sec
71:37:14	Burn to lower perigee to 125 n. mi.	CSM/LM	38	243	0:30
73:44:45	Out-of-plane burn with slight overspeed to raise apogee to 218 n. mi.	CSM/LM	3148	16 852	12:05
Total fuel consumed, lb				17 095	
Reserve, lb				260	

(c) LM-APS ΔV Budget

Initiation, hr:min:sec g.e.t.	Event	Vehicle configuration	ΔV, fps	Fuel used, lb	Duration, min:sec
120:42:25	Burn to raise apogee to 291 n. mi. and lower perigee to 117 n. mi. (FITH)	LM/APS	226	252	0:22
122:26:32	Burn to lower apogee to 195 n. mi. (CSI)	LM/APS	165	180	0:15
128:25:14	Burn to depletion	LM/APS	5552	4173	6:05
Total fuel consumed, lb				4605	
Reserve, lb				0	

TABLE I.- SPACECRAFT ΔV BUDGET - Continued

(d) SM RCS ΔV Budget

Initiation, hr:min:sec g.e.t.	SM RCS maneuvers	Maneuver attitude	Fuel used, lb	Total fuel for event, lb
3:38:52	Transposition (pullout)	+1 fps	6.7	
	Stop	-1 fps	6.7	
	Pitch 180° (2°/sec)		9.1	
	Roll 60°		4.14	
	Start back	+1 fps	6.7	
	Stop	-1 fps	6.7	
3:58:52	Soft docking		7.6	
			43.5	
4:43:52	CSM/LM separation from S-IVB	3 fps	33.4	76.9
	Platform alignment			
	Gross maneuver	1, 3 axis	9.52	
	Fine alignment	2, 2 axis	16.8	26.32
6:53:00	Nav. sightings			
	Gross maneuver (To pick up stars)	1, 3 axis	9.52	
	Establish pitch rate		1.00	
	Maintain local vertical	.2° deadband	.5	
	Change target	pitch & yaw	4.5	
	Establish pitch rate		1.0	
	Maintain local vertical		5	17.02
7:29:30	Orientation prior to first SPS burn	1, 3 axis	9.52	
7:31:32	1 SPS burn (54.7)			
	Roll control		.08	
	Shutdown transient		1.0	
	Switch wide to fine prior to burn		9.52	20.12
7:33:00	Nav. sightings (36 min.)			
	Gross maneuver	1, 3 axis	9.52	
	Establish pitch rate		1.00	
	Maintain local vertical	.2° deadband	.5	
	Change target	pitch & yaw	4.5	
	Establish pitch rate		1.00	
	Maintain local vertical	.2° deadband	.5	17.02

TABLE I - SPACECRAFT ΔV BUDGET - Continued

(d) SM RCS ΔV Budget - Continued

Initiation, hr:min:sec g.e.t.	SM RCS maneuvers	Maneuver attitude	Fuel used, lb	Total fuel for event, lb
8:20:00	Platform alignment			
	Gross maneuver	1, 3 axis	9.52	
	Fine alignment	2, 2 axis	16.8	
	Switch wide to fine deadband	1, 3 axis	9.52	
	Orientation	1, 3 axis	9.52	
	Fine deadband hold	.5°	.03	45.39
8:52:48	2nd SPS burn			
	Roll control		4.64	
	Shutdown transient		1.0	5.64
9-72	(Maximum deadband hold for 63 hours) (could just sleep)		15.	
72-74	(Deadband hold during first and second DPS burn		1.0	16.0
74-120	Maximum deadband hold 2.6 #/hr for fine DBD hold 1.2 lb/hr for wid. deadband hold		10.0	10.0
120:00:00	Platform alignment			
	Prior to separation			
	Gross maneuver	1, 3 axis	1.8	
	Fine alignment	2, 3 axis	3.6	
	Fine deadband hold for separation			5.4
	watching LM while APS (fifth)			
	sextant sighting on descent			
	Fine deadband hold during TPF			
	Sighting during TPF			
	Fine attitude hold during docking		20.0	20.
	CSM separation from APS	1 fps	4.3	
	Sighting on APS during burn	2 rolls	1.0	5.3
136:34:59	Platform alignment			
	Gross maneuver	1, 3 axis	1.8	
	Fine alignment	2, 3 axis	3.6	
	Orientation	1, 3 axis	1.8	
28:015:39	Ullage 2 jet		15.0	22.2

TABLE I.- SPACECRAFT ΔV BUDGET - Continued

(d) SM RCS ΔV Budget - Continued

Initiation, hr:min:sec g.e.t.	SM RCS maneuvers	Maneuver attitude	Fuel used, lb	Total fuel for event, lb
137:4:59	Third SPS burn (phasing)			
	Roll control		.5	
	Shutdown transient		1.0	
	Switch wide to fine		1.8	3.3
137-145	Maximum deadband hold	5°/sec	2.0	2.0
145:14:58	Platform alignment			
	Gross maneuver	1, 3 axis	1.8	
	Fine alignment	2, 2 axis	3.6	
	Orientation	1, 3 axis	1.8	
	Ullage 2 jets	1, 3 axis	15.0	22.2
145:45:58	4th SPS burn (CSI)			
	Roll control		0.0	
	Shutdown transient		1.0	1.0
	Sextant sighting (10)	10 rolls	3.0	
	Switch wide to fine		1.8	4.8
147:3:41	Platform alignment			
	Fine alignment	2, 2 axis	3.6	
	Orientation	1, 3 axis	1.8	
147:18:41	RCS burn (TPI)	10.3 fps	32.45	
	Attitude control		.1	
	Fine deadband hold 1.0 hr		2.6	40.55
149:00:00	Platform alignment			
	Fine alignment	2, 2 axis	3.6	
	Orientation	1, 3 axis	1.8	
	Ullage (2)	2 jets	.15	
149:31:00	5 SPS burn (TEI)			
	Switch to fine deadband		1.8	
149:33:00	Roll control		2.0	
	Shutdown transient		1.0	25.20
149-215	Maximum:			
	Deadband hold	5°/sec	13.	
	(small attitude changes for experiments)		12.	25.00
215:00:00	Platform alignment			
	2.0 Gross maneuver	1, 3 axis	1.8	
	Fine alignment	2, 3 axis	3.6	
	1.7 Orientation	1, 3 axis	1.0	
	Ullage	2 jets	15.0	21.40
				454.16
215:31:00	Deorbit burn			

TABLE I.- SPACECRAFT ΔV BUDGET - Continued

(d) SM RCS ΔV Budget - Concluded

Initiation, hr:min:sec g.e.t.	SM RCS maneuvers	Maneuver attitude	Fuel used, lb	Total fuel for event, lb
	SM spinup		8.0	
	SM jettison		14.0	
	RCS deorbit (153 fps)	minimum	550	
	The redline in reference 20 takes 751-lb total fuel		Used, nom redline	454.16 751.00 1205.16
			Margin	18.84

TABLE I .- SPACECRAFT ΔV BUDGET - Continued

(e) LM RCS ΔV Budget

g.e.t. hr:min:sec	Event	Vehicle configuration	Fuel used, lb
71:07:14	Platform alignment at .5 deg/sec	CSM/LM/DPS	- -
	Gross alignment (1-3 axis)	CSM/LM/DPS	8.8
	Coarse alignment (2-2 axis)	CSM/LM/DPS	16.8
	Fine alignment (2-2 axis)	CSM/LM/DPS	16.8
	Orientation (1-3 axis)	CSM/LM/DPS	8.8
	Switch to FDB (.3°) (1-3 axis)	CSM/LM/DPS	8.8
	DPS start transient	CSM/LM/DPS	10.0
	Ullage for 1st DPS burn (4 jets for 6 sec)	CSM/LM/DPS	8.6
71:37:14	Attitude control during 30 sec DPS burn	CSM/LM/DPS	2.1
	DPS shutdown transient	CSM/LM/DPS	0.6
	WDB hold (5.°) for 1.6 hr	CSM/LM/DPS	0.3
73:14:45	Platform alignment at .5 deg/sec	CSM/LM/DPS	- -
	Gross alignment (1-3 axis)	CSM/LM/DPS	8.8
	Fine alignment (2-2 axis)	CSM/LM/DPS	16.8
	Orientation (1-3 axis)	CSM/LM/DPS	8.8
	Switch to FDB (.3°) (1-3 axis)	CSM/LM/DPS	8.8
	DPS start transient	CSM/LM/DPS	1.0
	Ullage 2nd DPS burn (4 jets for 6 sec)	CSM/LM/DPS	8.6
73:44:45	Attitude control during 725 sec DPS burn	CSM/LM/DPS	7.2
120:30:00	LM separation from CSM (1 fps)	LM/DPS	1.8
	Platform alignment at 3. deg/sec	LM/DPS	- -
	Fine alignment (2-2 axis)	LM/DPS	2.0
	Orientation (1-3 axis)	LM/DPS	1.3
	Switch to FDB (.3°) (1-3 axis)	LM/DPS	1.3
	Ullage for 1st APS burn (4 jets for 3.5 sec)	LM/APS	5.0
120:42:25	Attitude (moment) control during 22 sec APS burn	LM/APS	10.2

TABLE I.- SPACECRAFT ΔV BUDGET - Concluded

(e) LM RCS ΔV Budget - Concluded

g.e.t. hr:min:sec	Event	Vehicle configuration	Fuel used, lb
	WDB hold (5.°) for 1.5 hr	LM/APS	0.5
122:11:32	Platform alignment at 3. deg/sec	LM/APS	- -
	Gross alignment (1-3 axis)	LM/APS	1.3
	Fine alignment (2-2 axis)	LM/APS	2.0
	Orientation (1-3 axis)	LM/APS	1.3
	Switch to FDB (.3°) (1-3 axis)	LM/APS	1.3
	Ullage for 2nd APS burn (4 jets for 3.5 sec)	LM/APS	5.0
122:26:32	Attitude (moment) control during 15 sec APS burn	LM/APS	10.2
	WDB hold (5.°) for 1.3 hr	LM/APS	0.5
123:40:57	Platform alignment at 3. deg/sec	LM/APS	- -
	Fine alignment (2-2 axis)	LM/APS	2.0
	Orientation (1-3 axis)	LM/APS	1.3
	Switch to FDB (.3°) (1-3 axis)	LM/APS	1.3
123:55:57	RCS burn for 26 sec (4 jet)	LM/APS	38.0
	WDB hold (5.°) for 1.0 hr	LM/APS	0.4
124:55:19	Platform alignment at 3. deg/sec	LM/APS	- -
	Fine alignment (2-2 axis)	LM/APS	2.0
	Orientation (1-3 axis)	LM/APS	1.3
	Switch to FDB (.3°) (1-3 axis)	LM/APS	1.3
125:10:19	RCS burn for 20 sec (4 jet)	LM/APS	29.0
	FDB hold (.3°) for .5 hr	LM/APS	3.9
125:27:00	Platform alignment at 3. deg/sec	LM/APS	- -
	Fine alignment (2-2 axis)	LM/APS	2.0
	Orientation (1-3 axis)	LM/APS	1.3
125:42:00	RCS burns for terminal rendezvous rendezvous (4 jet)	LM/APS	57.0
126:35:00	Hard docking	LM/APS	60.0
	Attitude control during APS burn to depletion	LM/APS	11.0
Total fuel used, lb			397.1

TABLE II. - AERODYNAMIC COEFFICIENTS AT TRIM ANGLE OF ATTACK AS A FUNCTION OF MACH NUMBER

M	α_{trim}	C_L trim	C_D trim	L/D_{trim}
0.2	174.47	.12947	.84682	.15289
0.4	167.20	.25715	.86124	.29857
0.7	165.31	.21798	.94879	.22974
0.9	160.97	.31036	1.0470	.29652
1.1	155.95	.45729	1.1859	.38560
1.2	154.55	.49591	1.2131	.40880
1.35	153.40	.55037	1.2753	.43157
1.65	153.27	.54235	1.2779	.42442
2.0	153.16	.52442	1.2759	.41103
2.4	153.68	.49944	1.2351	.40438
3.0	154.64	.48849	1.2096	.40385
6.0	156.64	.43103	1.2202	.35326
25.0	156.64	.43103	1.2202	.35326

TABLE III.- PARACHUTE DATA

(a) Drogue Chutes

Event	Time after 23 500-ft altitude, sec	C_DS per chute[a], ft^2
High altitude baroswitch closed	0.0	0
Deploy drogue chutes	2.0	0
Drogue chute reefed inflation	2.8	40
Drogue chute discrete	10.8	40
Drogue chutes full open	11.0	68

(b) Main Chutes

Event	Time after 10 200-ft altitude, sec	C_DS per chute[a], ft^2
Low altitude baroswitch closed	0.0	0
Main chute line stretch	2.0	0
Reefed inflation	3.5	300
Begin main chute disreef	10.0	300
Disreefing	12.5	2150
Main chutes full open	15.0	4000

[a]Drag of the command module is not included. Two drogues and three main parachutes are deployed.

TABLE IV.- SPACECRAFT GUIDANCE USED FOR EACH BURN

Event No.	Event description and active module	Time of initiation, hr:min:sec g.e.t.	Guidance used	Comment
8	Midcourse correction burn	7:31:32.3	external delta V	in-plane burn
11	SPS burn to circularize	8:52:48	Lambert	in-plane burn
13	1st docked DPS burn to lower perigee	71:37:14	external delta V	in-plane burn
14	2nd docked DPS burn	73:44:45	external delta V	out-of-plane
16	Separation (LM/RCS)	120:30:0	external delta V	
17	5th burn (LM/APS)	120:42:25	external delta V	
18	CSI burn (LM/APS)	122:26:32	external delta V	
19	CDH burn (LM/RCS)	123:55:57	external delta V	
20	TPI burn (LM/RCS)	125:11:59	external delta V	
21	TPF burns (2) (LM/RCS)	125:53:31	Lambert	
23	Separation (CSM/RCS)	128:00:00	external delta V	
24	Burn to depletion LM/APS	128:25:14	external delta V	out-of-plane
25	3rd SPS burn to get phasing with descent	137:04:59	external delta V	in-plane
26	4th SPS burn to get phasing with descent	145:45:58	external delta V	in-plane
27	CSM/RCS burn for TPI	147:18:27	external delta V	in-plane
29	5th SPS burn to simulated transearth	149:10:24	external delta V	out-of-plane
31	Deorbit burn (CSM/SPS)	238:41:24	external delta V	in-plane

TABLE V.- GROUND NETWORK UNIFIED S-BAND STATIONS

Call letters	Station name (USB)	Geodetic latitude, deg	Geodetic longitude, deg	Altitude (height above mean sea level), n. mi.	Slant range, n. mi.
HAW	Hawaii	22.12807249	-159.66722638	0.61388234	225000.0
MERCURY	Inject ship (1)	20.00000000	-135.00000000	0.00000000	225000.0
GST	Goldstone	35.34169500	-116.87328833	0.52145816	225000.0
GYM	Guaymas, Mex.	27.96320610	-110.72084916	0.01048532	225000.0
REDSTONE	Inject ship (2)	-15.00000000	10.00000000	0.00000000	225000.0
TEX	Corpus Christi	27.65361111	-97.37833333	0.00000000	225000.0
VANGUARD	Insertion ship	25.00000000	-50.00000000	0.00000000	225000.0
MILA	Merritt Island	28.50827222	-80.52674611	0.00672798	225000.0
GBI	Grand Bahama	26.65416666	-78.15277778	0.00000000	225000.0
BDA	Bermuda	32.35128694	-64.65818111	0.01133948	225000.0
ANT	Antigua	17.01944444	-61.75000000	0.00000000	225000.0
ASC	Ascension	-7.95523416	-14.32757889	0.30302419	225000.0
MAD	Madrid	40.45535833	-4.16739555	0.44559884	225000.0
CRO	Carnarvon	-24.90756278	113.72424722	0.03099348	225000.0
GUA	Guam	13.30823555	144.73441361	0.06916919	225000.0
CNB	Canberra	-35.59722222	148.97916666	0.00000000	225000.0
CYI	Grand Canary	27.740277	-15.604166	0.00000000	225000.0

TABLE VI.- GROUND NETWORK EQUIPMENT

Call letters	Station name	Type of Equipment Available					
		Unified S-band	C-band radar	UHF command	VHF telemetry (on-site)	VHF telemetry (MCC-H)	VHF voice
HAW	Hawaii	USB (30)	X	X	X	X	X
MERCURY	Inject ship (1)	USB (30)	X	X	X		X
GST	Goldstone	USB (85)					
GYM	Guaymas, Mexico	USB (30)			X	X	X
REDSTONE	Inject ship (2)	USB (30)	X	X	X		X
TEX	Corpus Christi	USB (30)		X	X	X	X
VANGUARD	Insertion ship	USB (30)	X	X	X		X
MILA	Merritt Island	USB (30)	X	X	X	X	X
GBI	Grand Bahama	USB (30)	X	X	X	X	X
BDA	Bermuda	USB (30)	X	X	X	X	X
ANT	Antigua	USB (30)	X	X	X	X	X
ASC	Ascension	USB (30)	X		X	X	X
MAD	Madrid	USB (85)					
CRO	Carnarvon	USB (30)	X	X	X	X	X
GUA	Guam	USB (30)			X	X	X
CNB	Canberra	USB (85)					
CYI	Grand Canary	USB (30)	X	X	X	X	X

TABLE VII.- SPACECRAFT RELATIVE ATTITUDES AT EVENT INITIATION

Event	Relative Attitudes[a], deg		
	Pitch	Yaw	Roll
(a) First Period of Activity, CSM			
S-IVB cutoff (event 4)	0.0	0.0	180.0 (headsdown)
Solar orientation (event 5)	-85.1	0.0	-174.8
First SPS burn (event 8)	0.0	0.0	0.0
Second SPS burn (event 11)	153.0	-7.2	176.3
(b) Second Period of Activity, CSM (CSM/LM docked)			
First DPS (event 13)	179.7	0.0	180.0 (headsdown)
Second DPS (event 14)	159.9	-87.5	159.9

Pitch = Angle between spacecraft X-axis and local horizontal plane

Yaw = Angle between spacecraft X-axis and the vertical plane containing the velocity vector

Roll = Angle between spacecraft Z-axis and the local vertical

TABLE VII.- SPACECRAFT RELATIVE ATTITUDES AT EVENT INITIATION - Continued

Event	Relative Attitudes[a], deg		
	Pitch	Yaw	Roll
(c) Third Period of Activities			
LM-CSM separation	179.9	0.004	-180.0
FITH APS burn	-62.9	0.0	0.0
Reorient for rendezvous radar	90.0	0.0	0.0
CSI APS burn	179.4	0.0	-180.0
CDH RCS burn	-54.5	0.0	0.0
Continuous pitch for radar lock-on	90.0	0.0	0.0
TPI RCS burn	32.3	0.9	-0.6
First TPF burn	-50.1	0.6	0.7
Second TPF burn	-29.6	-5.4	-3.1

TABLE VII.- SPACECRAFT RELATIVE ATTITUDES AT EVENT INITIATION - Concluded

Event	Relative Attitudes[a], deg		
	Pitch	Yaw	Roll
(d) Fourth and Fifth Period of Activities			
Third SPS burn	-121.0	-0.19	-179.7
Reorient for sextant sightings	57.0	0.0	0.0
Fourth SPS burn for CSI	-61.2	0.59	1.08
RCS burn for TPI	-152.9	-2.23	-178.9
Simulated transearth burn	168.2	89.99	-168.2
Deorbit burn	-130.7	0.0	-180.0

TABLE VIII.- TRANSPOSITION AND DOCKING

(a) CSM/LM/S-IVB Attitude Timeline

Time from S-IVB cutoff, min	ΔT, min	Event	Comment
0	15	1. S-IVB cutoff	S-IVB/IU/CSM/LM
		Establish zero degree attitude between spacecraft +X axis (forward) and local horizontal.	
		Establish and maintain orbital pitch rate.	
		S-IVB venting and blowdown.	
15	6.5	2. Move S-IVB to solar orientation for inactive vehicle illumination	Puts sun 30° off the S-IVB X-axis
		Pitch* (-103.5° at 0.3°/sec)	
		Roll (5.2° at 0.5°/sec)	
		*Time dependent	
21.5		3. Maintain inertial attitude ($\pm 1.^\circ$ deadband)	
22		4. +X transposition to 100 ft using CSM-RCS quads.	CSM
		Rate - 3 fps (10 sec RCS burn)	
26		ENTER SUNLIGHT	
	8.5	Orient CSM for docking	Crew has sun above and behind them.
		Pitch ($+180^\circ$ at 0.5°/sec)	
		Roll (-60° at 0.5°/sec)	
		+X transposition back to LM	
		Rate (> 50 ft) 1. fps avg.	
		Rate (< 50 ft) 0.1 to 1. fps	
	10.5	Accomplish soft docking	
	20	(Preliminary value)	
42	45	5. Hard dock	S-IVB/IU/CSM/LM

TABLE VIII.- TRANSPOSITION AND DOCKING - Continued

(a) CSM/LM/S-IVB Attitude Timeline - Concluded

Time from S-IVB cutoff, min	ΔT, min	Event	Comment
87	5	6. Extract LM from S-IVB	CSM/LM
		Rate (1.5 fps using RCS quads)	
		Orient LM forward in direction of travel	
120		7. Accomplish S-IVB post separation maneuver.	S-IVB
		Establish inertial attitude hold	
	20	Vent S-IVB to maximize the separation distance between the CSM/LM and the S-IVB.	

TABLE VIII.- TRANSPOSITION AND DOCKING - Concluded

(b) CSM T & D Attitudes at Maneuver Initiation

Event	Spacecraft relative attitude[a], deg			Launch site inertial attitude[b], deg		
	Pitch	Yaw	Roll	Pitch	Yaw	Roll
S-IVB cutoff	0.	0.	180.0	-160.9	0.2	-178.7
S-IVB solar orientation	0.	0.	180.0	137.2	-1.1	-179.2
Maintain inertial attitude	-85.1	0.	-174.8	33.7	-0.5	-176.0
Soft dock	121.0	0.	114.8	-146.3	0.5	116.0

[a]Pitch = Angle between spacecraft X-axis and local horizontal plane

Yaw = Angle between spacecraft X-axis and the vertical plane containing the velocity vector

Roll = Angle between spacecraft Z-axis and the local vertical

[b]Pitch = rotation about Y-axis
Yaw = rotation about Z-axis
Roll = rotation about X-axis.

TABLE IX .- MAJOR MISSION EVENTS

(a) First period of activities

Event number	Event description and active module	Time of initiation, hr:min:sec g.e.t.	Duration, min:sec	Position at initiation			Resulting perigee/ apogee, n. mi.	ΔV fps	Station tracking
				Lat., deg.	Long., deg.	Alt., n. mi.			
1	Launch	00:00:00	11:25	28.5	-80.6	---	---	---	MILA GBI BDA
2	Insertion into 100 n. mi. circular earth orbit	00:11:25	---	32	-54	104	100/104	---	Vanguard
3	Platform alignment	2:26:00	52:00	---	---	---	---	---	HAW US Networks
4	Injection into high apogee elliptic orbit (3950 n. mi.)	3:16:52	02:33	26	-62	104	107/3950	4220	BDA Vanguard ANT
5	Transposition, docking, and separation (CSM/RCS)	3:38:53	65:00	-31	52	2700	107/3950	---	CRO ASC Redstone HAW
6	Platform alignment	6:03:00	31:00	30	-120	200	107/3950	---	US Networks
7	Deep space navigation by crew	6:54:39	36:00	-31	14	3000	107/3950	---	ASC Redstone
8	Midcourse correction burn at apogee to raise perigee to 150 n. mi., 1st SPS burn	7:31:32	00:08	-24	57	3947	150/3950	55	CRO Redstone

TABLE IX .- MAJOR MISSION EVENTS - Continued

(a) First period of activities - Concluded

Event number	Event description and active module	Time of initiation hr:min:sec g.e.t.	Duration, hr:min:sec	Position at initiation			Resulting perigee/ apogee n. mi.	ΔV, fps	Station tracking
				Lat., deg	Long., deg	Alt., n. mi.			
9	Deep space navigation by spacecraft crew	7:38:30	0:36:00	-24	57	3947	150/3950	---	CRO GUA Redstone
10	Platform alignment	8:20:00	0:31:00	5	104	2000	150/3950	---	GUA
11	Second SPS burn to circularize into 150 x 200 n. mi.	8:52:48	0:07:52	31	-160	192	150/200	4069	HAW Mercury
12	Simulated translunar coast	9:00:41	62:30:33	19	-126	148	150/200	---	All Networks
(b) Second period of activities									
13	Docked DPS burn to lower perigee (CSM + LM/DPS)	71:37:14	00:00:30	-28	150	189	125/192	38	CNB
14	Docked DPS burn out-of-plane (CSM + LM/DPS)	73:44:45	00:12:05	32	-100	138	127/218	3148	TEX MILA GBI BDA Vanguard
15	Simulated lunar orbit coast	73:56:50	46:33:10	25	-48	127	127/218	---	All Networks

TABLE IX .- MAJOR MISSION EVENTS - Continued

(c) Third period of activities

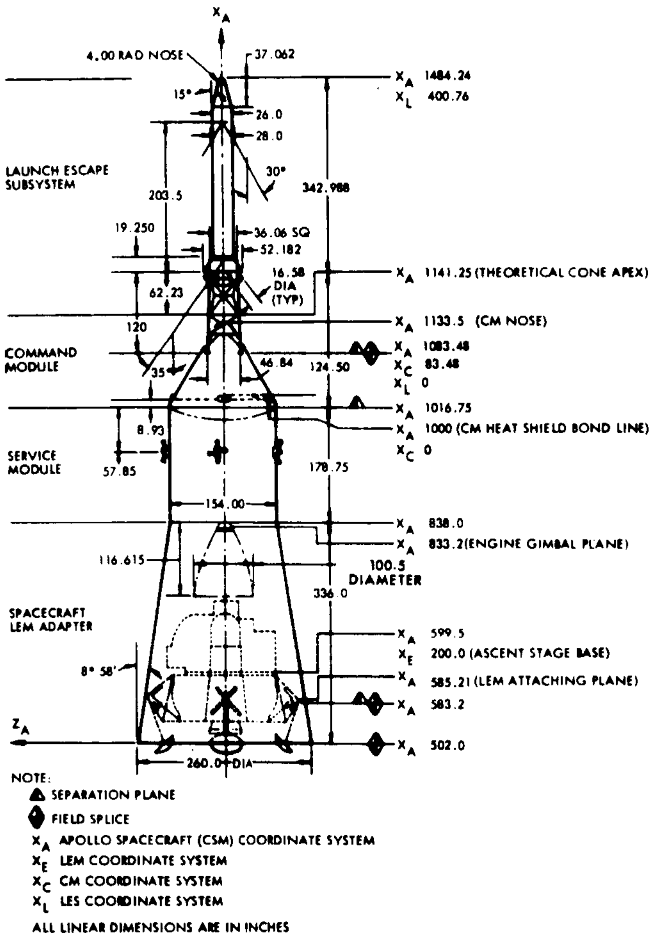
Event number	Event description and active module	Time of initiation, hr:min:sec g.e.t.	Duration, min:sec	Position at initiation			Resulting perigee/ apogee, n. mi.	ΔV, fps	Stations tracking
				Lat., deg	Long., deg	Alt., n. mi.			
16	Separation (LM/RCS)	120:30:00	00:01	19	-135	180	128/217	1	Mercury
17	FITH burn (LM/APS)	120:42:25	00:22	31	-84	144	117/291	226	TEX MILA GBI
18	CSI burn (LM/APS)	122:26:32	00:15	20	-57	117	116/195	164	ANT Vanguard
19	CDH burn (LM/RCS)	123:55:57	00:26	21	-83	116	116/206	32	MILA GBI
20	TPI burn (LM/RCS)	125:10:19	00:20	29	-174	146	121/216	24	HAW
21	Braking (2 burns) (LM/RCS)	125:42:00 to 125:50:33	08:33	-10	-53	147	128/217	21	None
22	Soft and hard docking (LM/RCS)	126:06:00 to 126:35:00	29:00	-27	41	213	128/217	25	None
23	Separation (CSM/RCS)	128:00:00	00:02	12	96	194	128/217	1	None
24	Burn to depletion (LM/APS)	128:25:14	06:05	25	-162	130	128/234	5552	HAW Mercury

TABLE IX .- MAJOR MISSION EVENTS - Concluded

Event number	Event description and active module	Time of initiation, hr:min:sec g.e.t.	Duration, hr:min:sec	Position at initation			Resulting perigee/ apogee, n. mi.	ΔV, fps	Stations tracking
				Lat., deg	Long., deg	Alt., n. mi.			
(d) Fourth period of activities									
25	Third SPS burn to obtain proper phasing with descent stage	137:04:59	00:00:03	13	-42	196	109/219	60	None
26	Fourth SPS burn for corrective maneuver to rendezvous with descent stage	145:45:58	00:00:03	-26	108	209	113/215	77	CRO
27	CSM/RCS burn for flyby (TPI)	147:18:27	00:00:22	-24	91	211	129/215	10	None
28	Flyby	148:00:00	00:05:00	28	-118	141	129/215	---	GYM, GST, TEX
29	Fifth SPS burn to simulate transearth burn	149:10:24	00:00:29	15	137	194	93/200	710	GUA
30	Simulated trans-earth coast	149:10:53	89:38:03	16	138	193	93/200	---	All Networks
(e) Fifth period of activities									
31	Deorbit burn into AMR (CSM/SPS)	238:41:24	00:14	6	-177	186	-20/186	370	Watertown (Entry ship)
32	CM/SM separation	238:59:07	06:31	29	-110	99	---	408	GYM
33	Reentry	239:04:07	17:00	29	-90	66	---	---	TEX
34	Landing	239:21:07	---	21.3	-58.6	0.0	---	---	ANT

TABLE X.- REENTRY PARAMETERS

Inertial velocity, fps	25 750.19
Inertial flight-path angle, deg	-1.65
Inertial azimuth, deg	96.27
Altitude, ft	400 000
Entry latitude, deg	28.62
Entry longitude, deg	89.53
Time to 400 000 ft, hr g.e.t.	239.068



(a) Apollo spacecraft reference dimensions.

Figure 1. - Vehicle configuration and coordinate systems.

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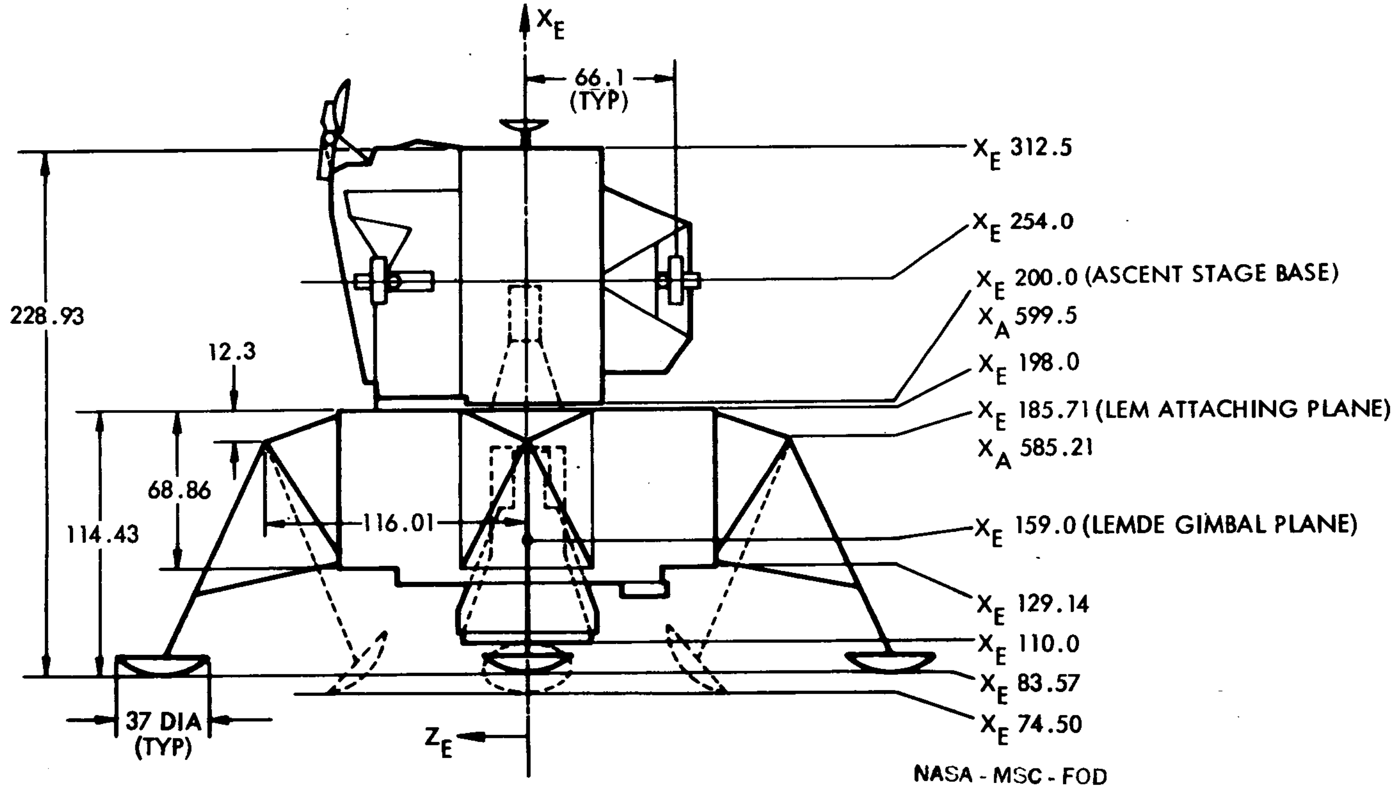

NOTE: X_E = LEM COORDINATE SYSTEM
X_A = APOLLO SPACECRAFT COORDINATE SYSTEM
ALL LINEAR DIMENSIONS ARE IN INCHES
RCS PITCH AND ROLL JETS Z_E AND Y_E (EFFECTIVE GEOMETRICAL MOMENT ARM) = $\pm$ 66.1

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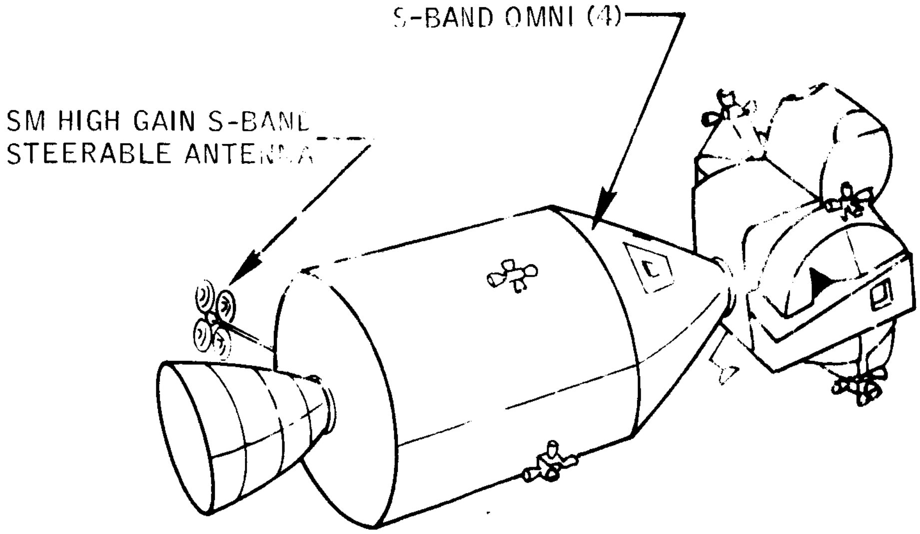
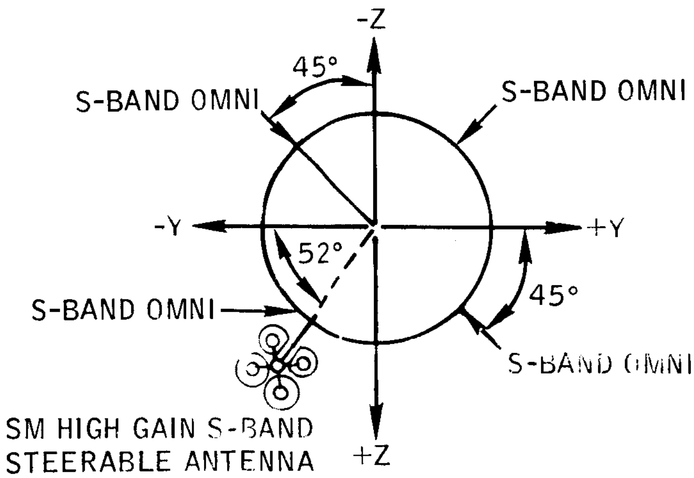
Date 5-13-66 DBGrammer

(b) LM reference dimensions.

Figure 1. - Continued.

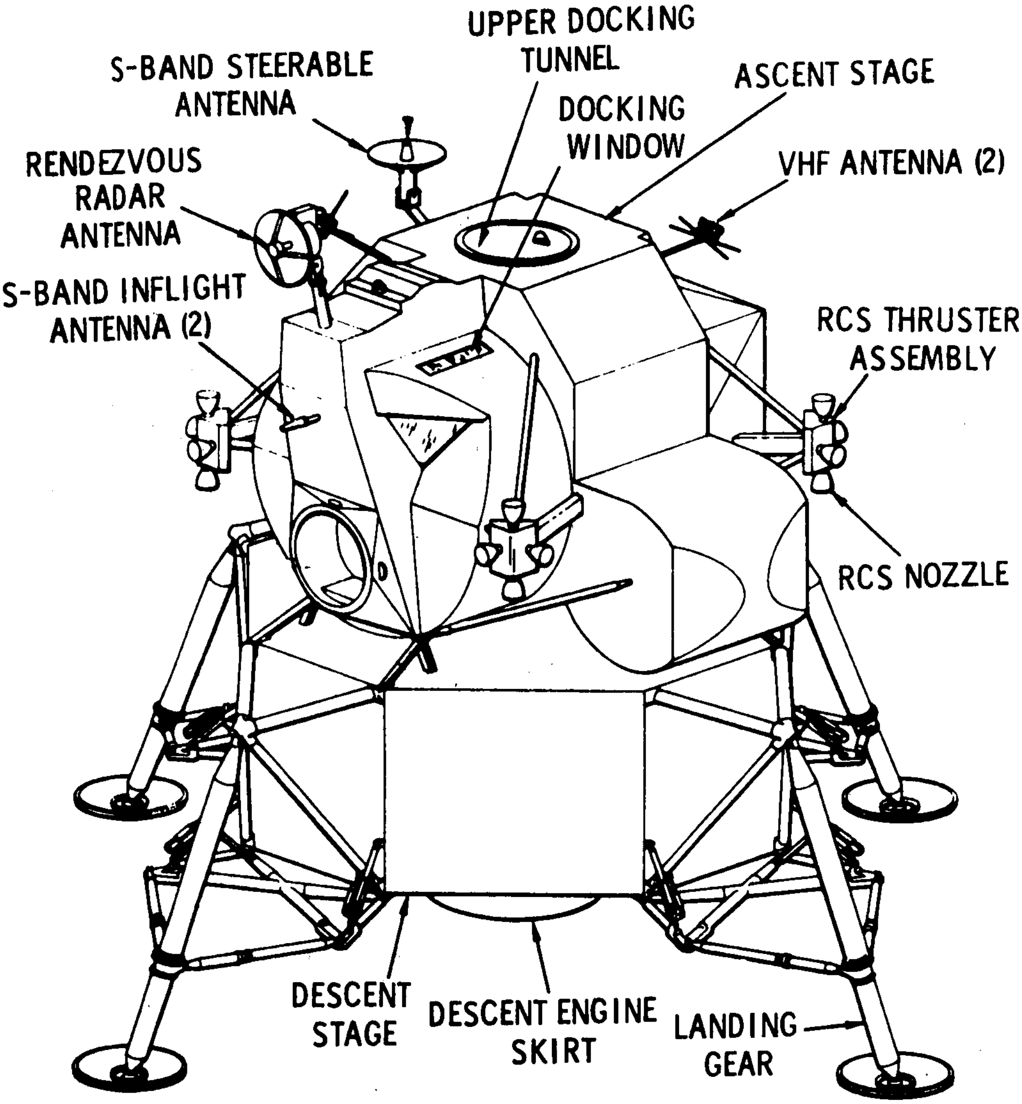




CSM BODY AXIS SYSTEM

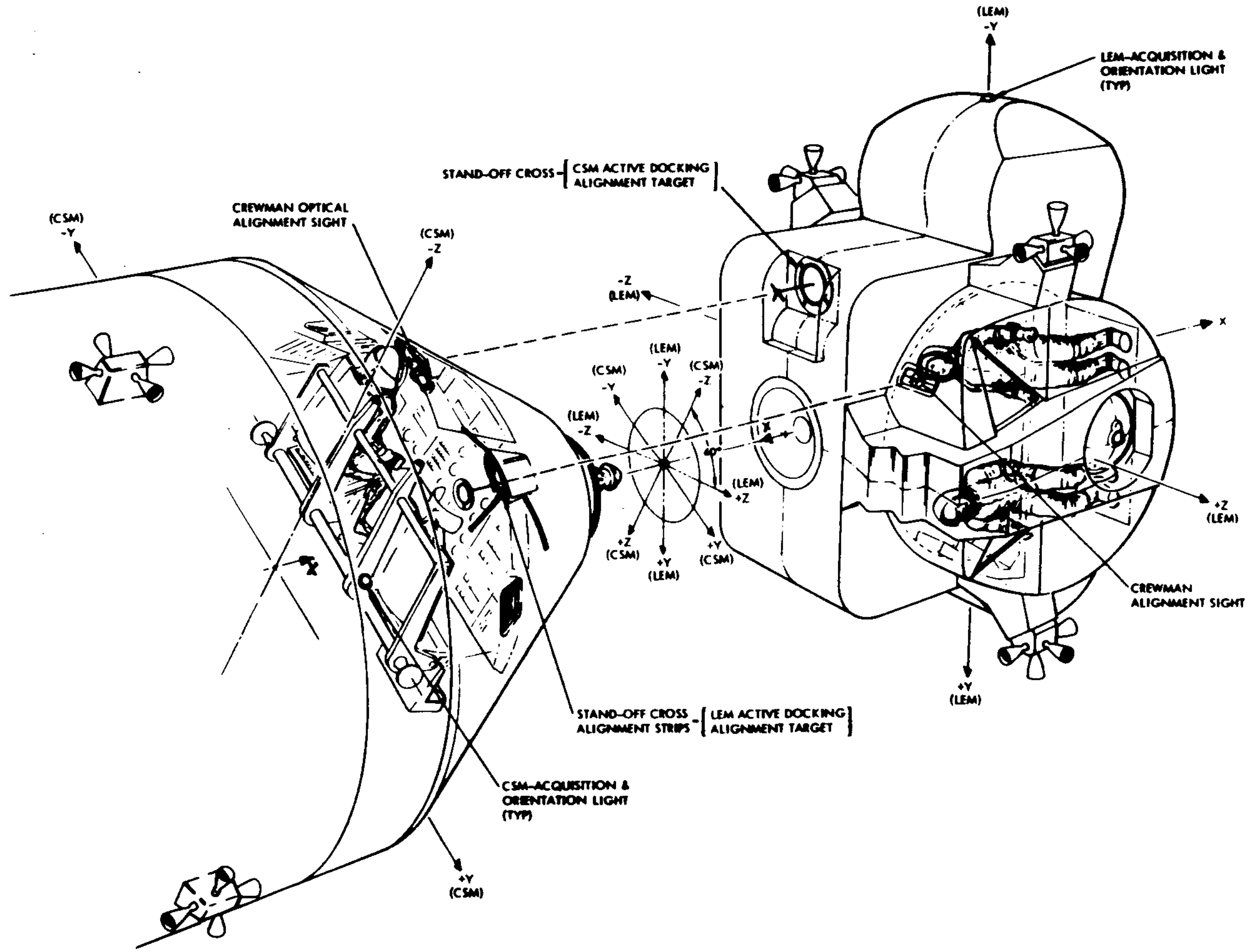
(c) CSM/LM docked configuration antenna locations, block 2.

Figure 1. - Continued.



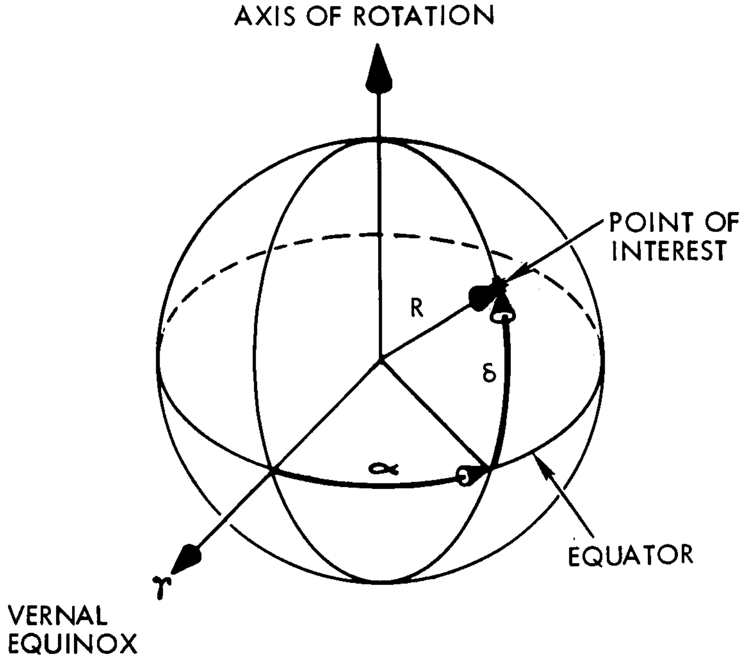
(d) LM outboard profile.

Figure 1. - Continued.



(e) Docked CSM/LM relative body coordinate system.

Figure 1. - Continued.



TYPE: Non-rotating, earth referenced

ORIGIN: The center of the earth

ORIENTATION AND DEFINITION:

The point of interest is defined by a radius (R), its right ascension (α), and declination (δ).

The radius is the radial distance from the center of the earth to the point of interest.

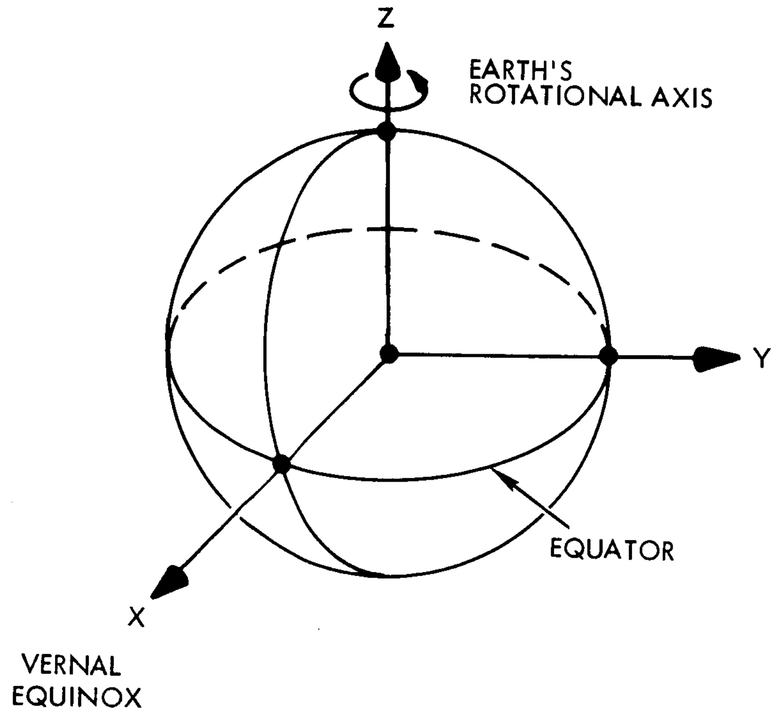
The right ascension is the angle measured from the vernal equinox, eastward, along the equator, to the meridian which passes through the point of interest.

The declination is the angle between the radius vector (R) and the equatorial plane.

This system also defines the selenocentric system when translated along the earth-moon line through the radial distance of the moon from the earth. The same epoch and equinox-equator flexibility as described for the geocentric inertial coordinate system applies for this coordinate system.

(f) Geocentric polar coordinate system.

Figure 1. - Continued.



TYPE: Earth referenced, non-rotating

ORIGIN: The center of the earth.

ORIENTATION AND DEFINITION:

The Z-axis is directed along the earth's rotational axis, positive north.

The X-axis is directed toward the vernal equinox.

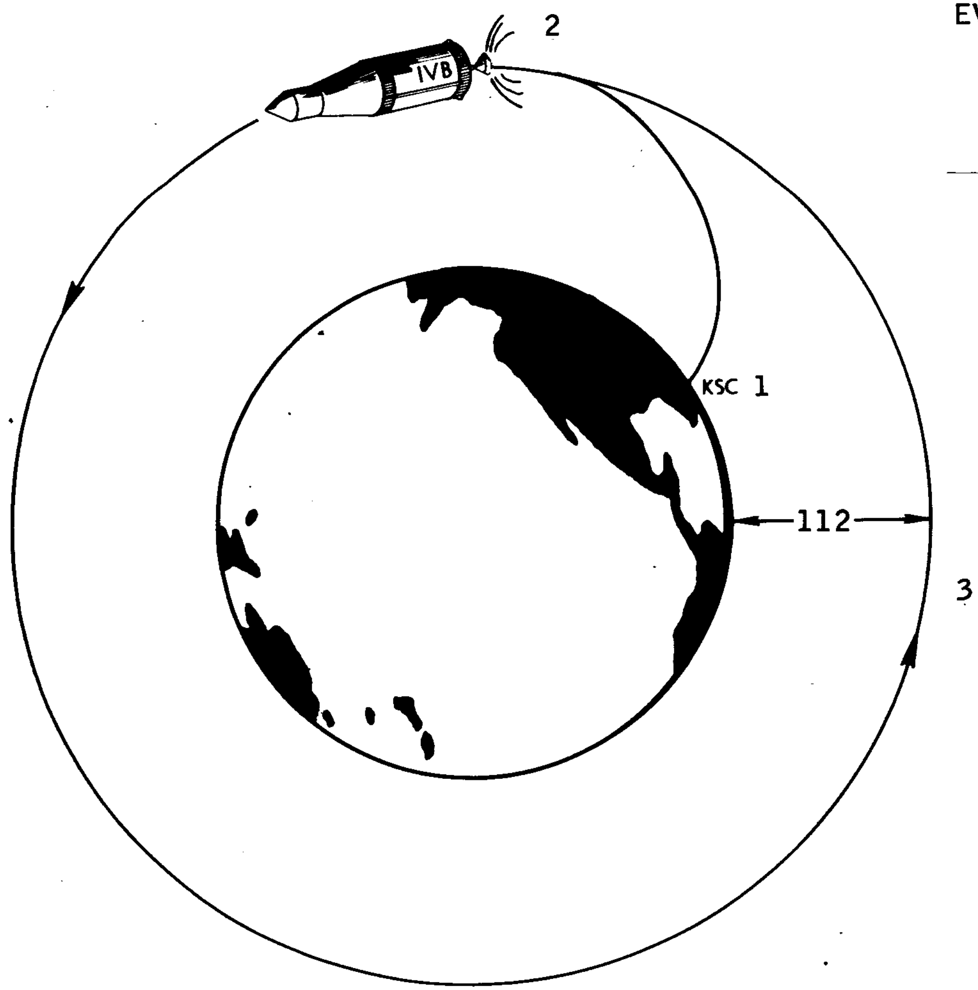
The Y-axis completes the standard right-handed system.

The epoch will generally be the nearest beginning of a Besselian year (nearest to initial trajectory computation time). However, special applications may involve other epochs. The equator and equinox can be either mean-of-epoch or true-of-date.

This system also defines the selenocentric system when translated along the earth-moon line through the radial distance of the moon from the earth.

(g) Geocentric inertial coordinate system.

Figure 1. - Concluded.

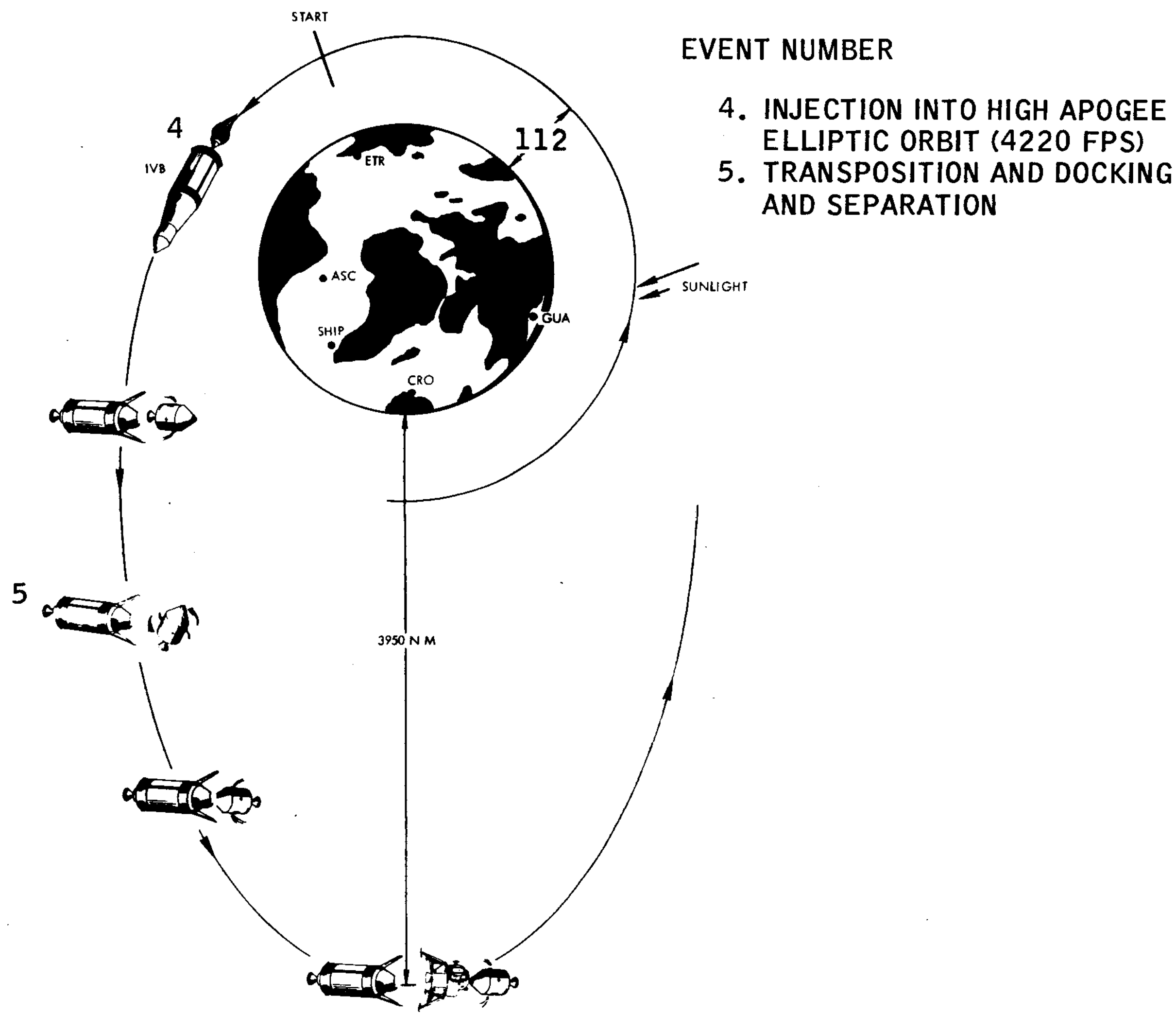


EVENT NUMBER

1. LAUNCH
2. INSERTION
3. PLATFORM ALIGNMENT

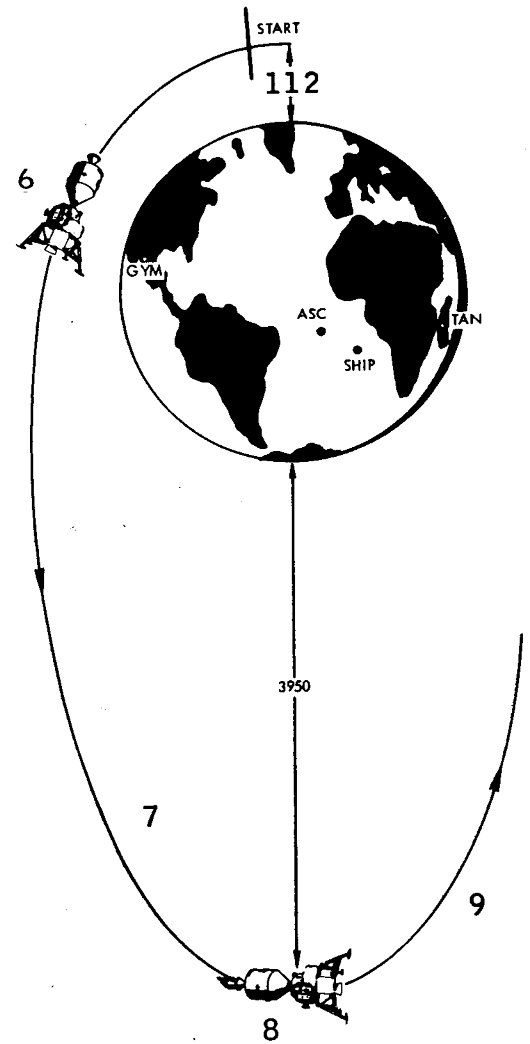
(a) Diagram of events 1 through 3.

Figure 2.- Events of first period of activity.



(b) Diagram of events 4 through 5.

Figure 2.- Continued.

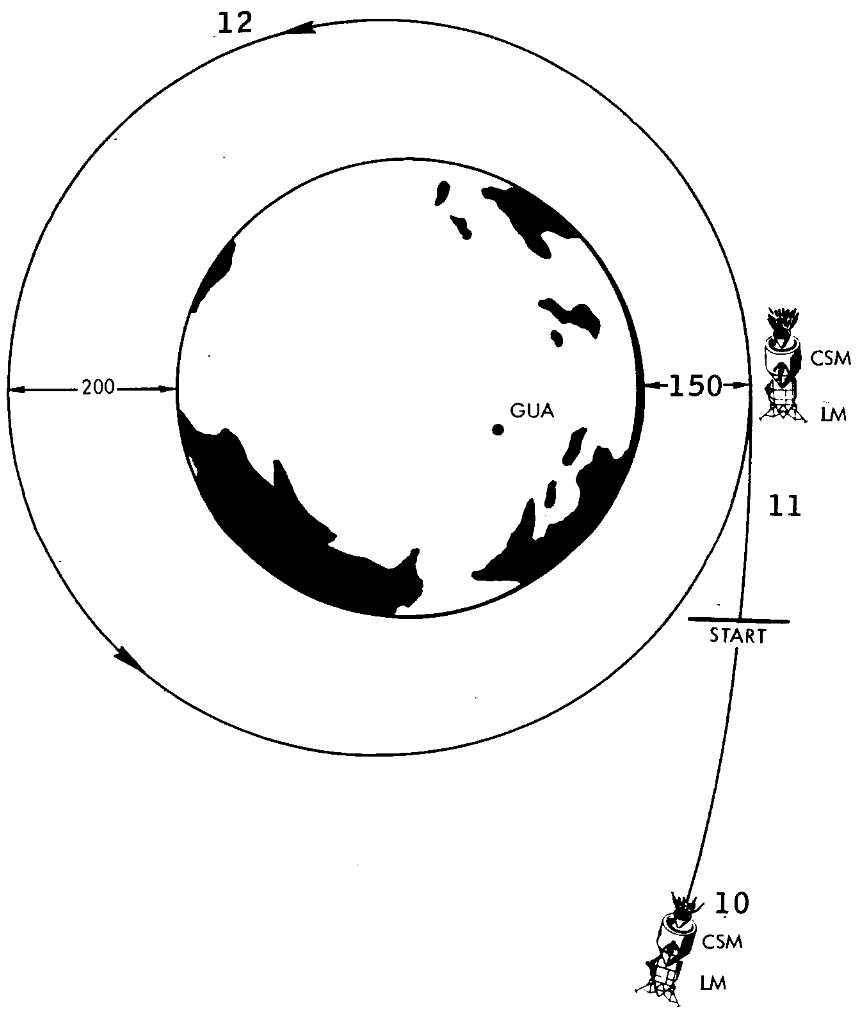


EVENT NUMBER

6. PLATFORM ALIGNMENT
7. DEEP SPACE NAVIGATION
8. MIDCOURSE CORRECTION BURN (54 FPS)
9. DEEP SPACE NAVIGATION

(c) Diagram of events 6 through 9.

Figure 2.- Continued.

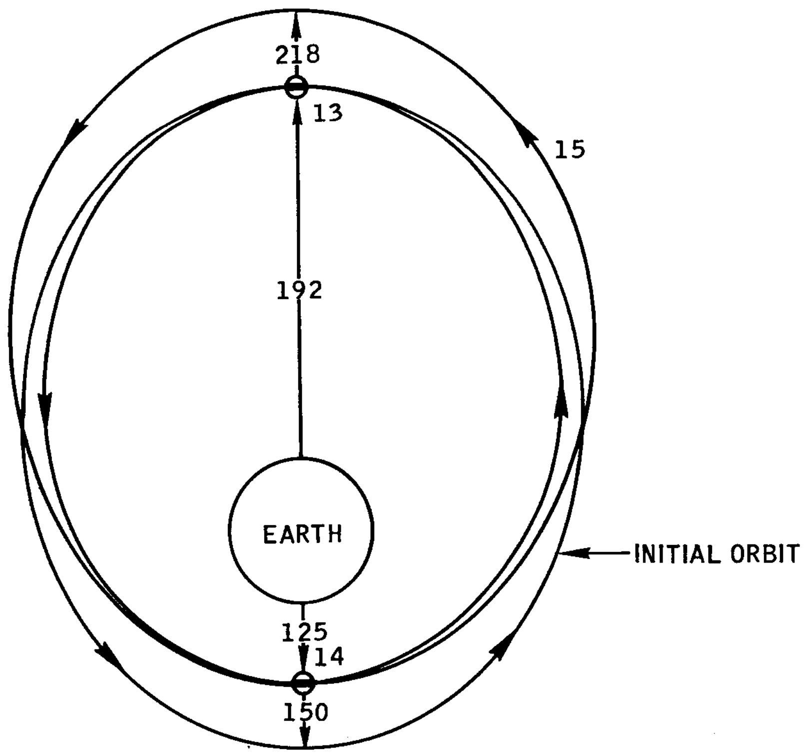


EVENT NUMBER

10. PLATFORM ALIGNMENT
11. SECOND SPS BURN TO CIRCULARIZE INTO 150 X 200 N.MI.
12. SIMULATED TRANSLUNAR COAST

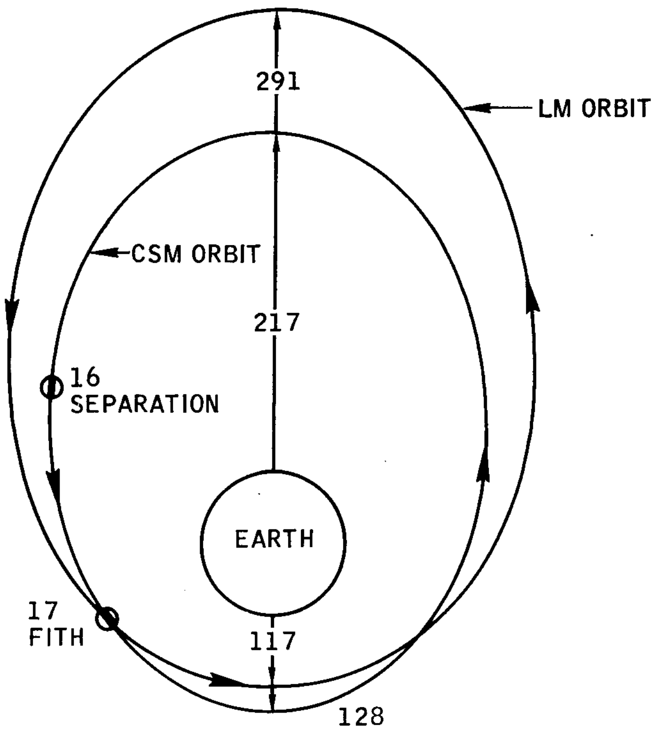
(d) Diagram of events 10 through 12.

Figure 2.- Concluded.

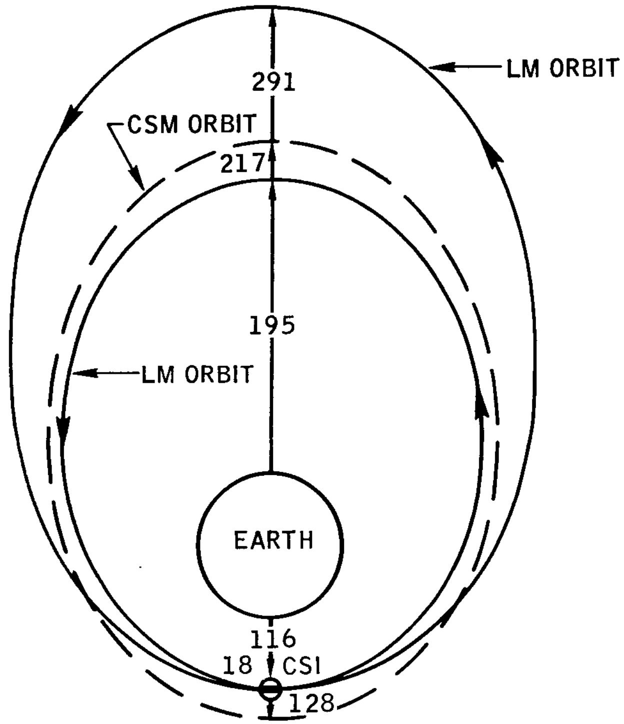


13. DOCKED CSM/LM/DPS BURN TO LOWER PERIGEE TO 125 N. MI.
$\Delta V \approx 38$ FPS
14. DOCKED CSM/LM/DPS BURN (MOSTLY OUT-OF-PLANE) TO RAISE APOGEE TO 218 N. MI.
$\Delta V \approx 3148$ FPS
15. SIMULATED LUNAR COAST FOR 46.5 HR

Figure 3.- Diagram of events 13 through 15, second period of activity.



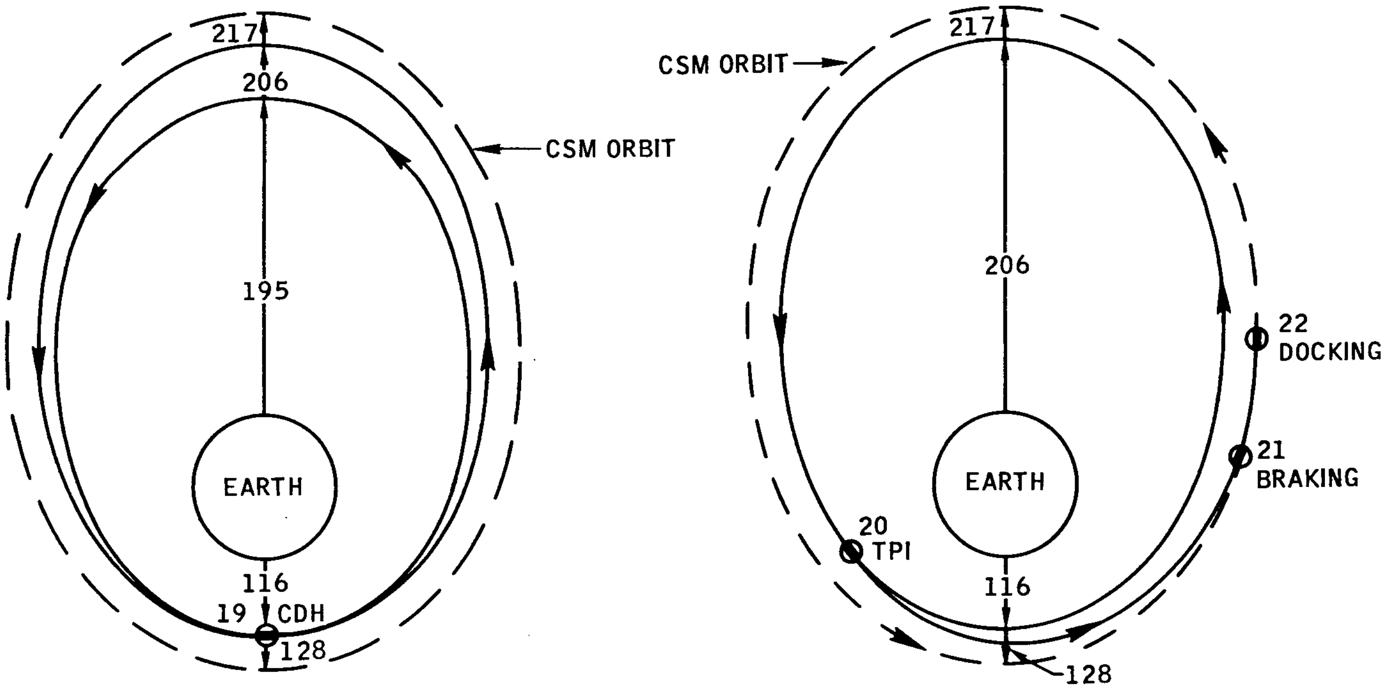
16. LM SEPARATION FROM CSM
$\Delta V \approx 1$FPS
17. LM/APS BURN (FITH) RAISES APOGEE TO 291 N. MI. AND LOWERS PERIGEE TO 117 N. MI.
$\Delta V \approx 226$ FPS



18. LM/APS BURN (CSI) LOWERS APOGEE TO 195 N. MI.
$\Delta V \approx 165$ FPS

(a) Diagram of events 16 through 18.

Figure 4.- Events of third period of activity.



19. LM/RCS BURN (CDH) RAISES APOGEE TO 206 N. MI.
$\Delta V \approx 32$ FPS

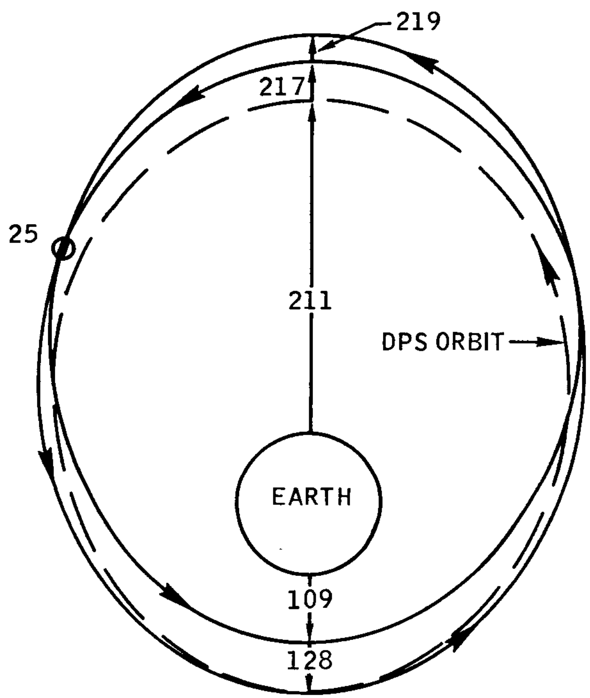
20. LM/RCS BURN (TPI) TO INTERCEPT CSM
$\Delta V \approx 24$ FPS

21. LM/RCS BURNS (BRAKING) FOR TERMINAL RENDEZVOUS
$\Delta V \approx 21$ FPS

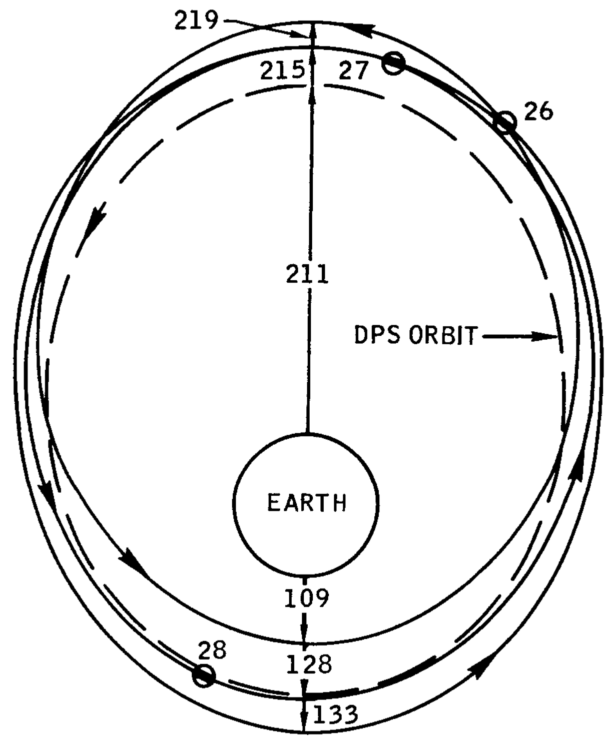
22. LM/RCS BURNS FOR SOFT AND HARD DOCKING
$\Delta V \approx 25$ FPS

(b) Diagram of events 19 through 22.

Figure 4.- Concluded.



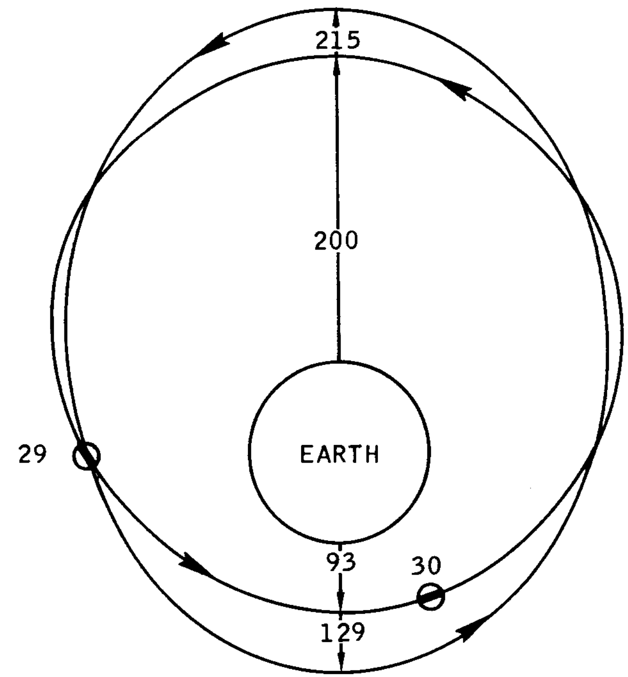
25. SPS BURN INTO 109-BY 219-N. MI. ORBIT TO OBTAIN PROPER PHASING WITH DPS
$\Delta V \approx 59$ FPS



26. SPS BURN INTO 133-BY 215-N. MI. ORBIT TO SETUP TPI BURN
$\Delta V \approx 77$ FPS
27. RCS BURN TO LOWER PERIGEE TO 129 N. MI. (TPI)
$\Delta V \approx 10$ FPS
28. FLY-BY

(a) Diagram of events 25 through 28.

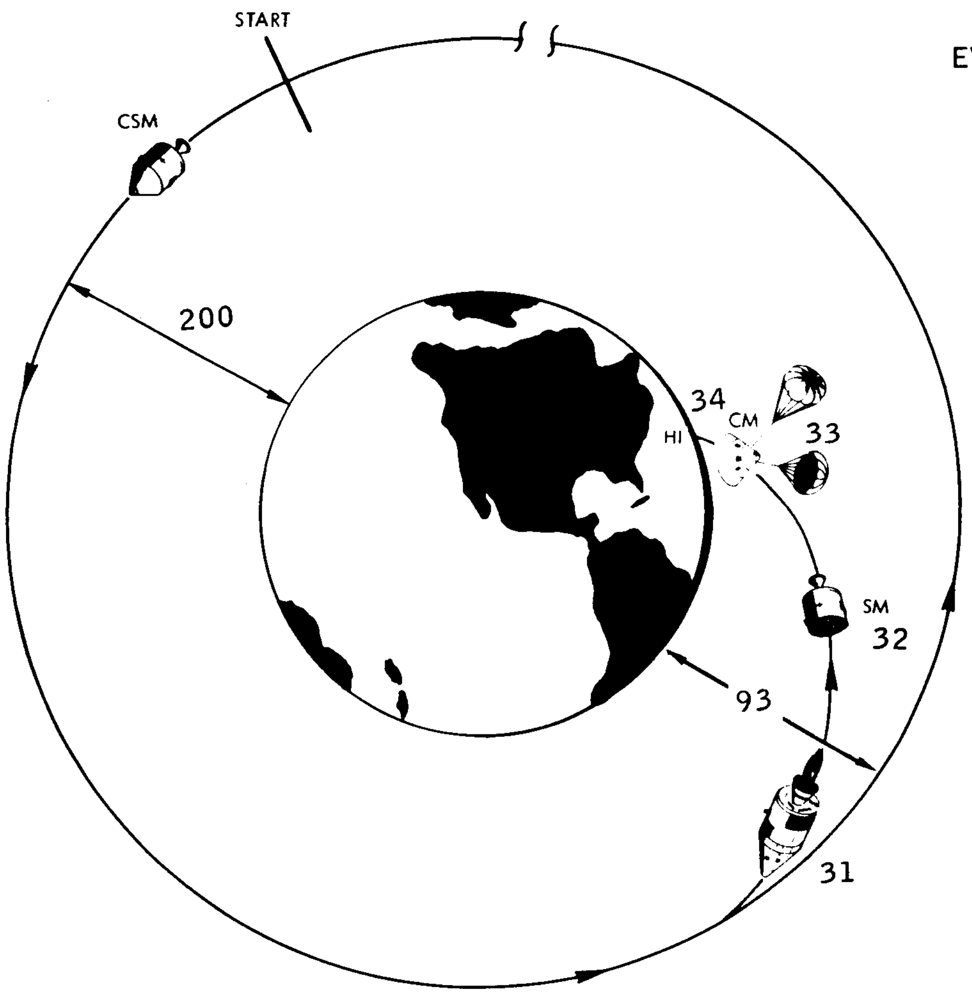
Figure 5.- Events of fourth period of activity.



29. SPS BURN LOWERS PERIGEE TO 93 N. MI. AND APOGEE TO 200 N. MI. (TEI SIMULATION) $\Delta V \approx 710$ FPS
30. SIMULATED TRANSEARTH COAST FOR 89.5 HOURS

(b) Diagram of events 29 through 30.

Figure 5.- Concluded.



EVENT NUMBER

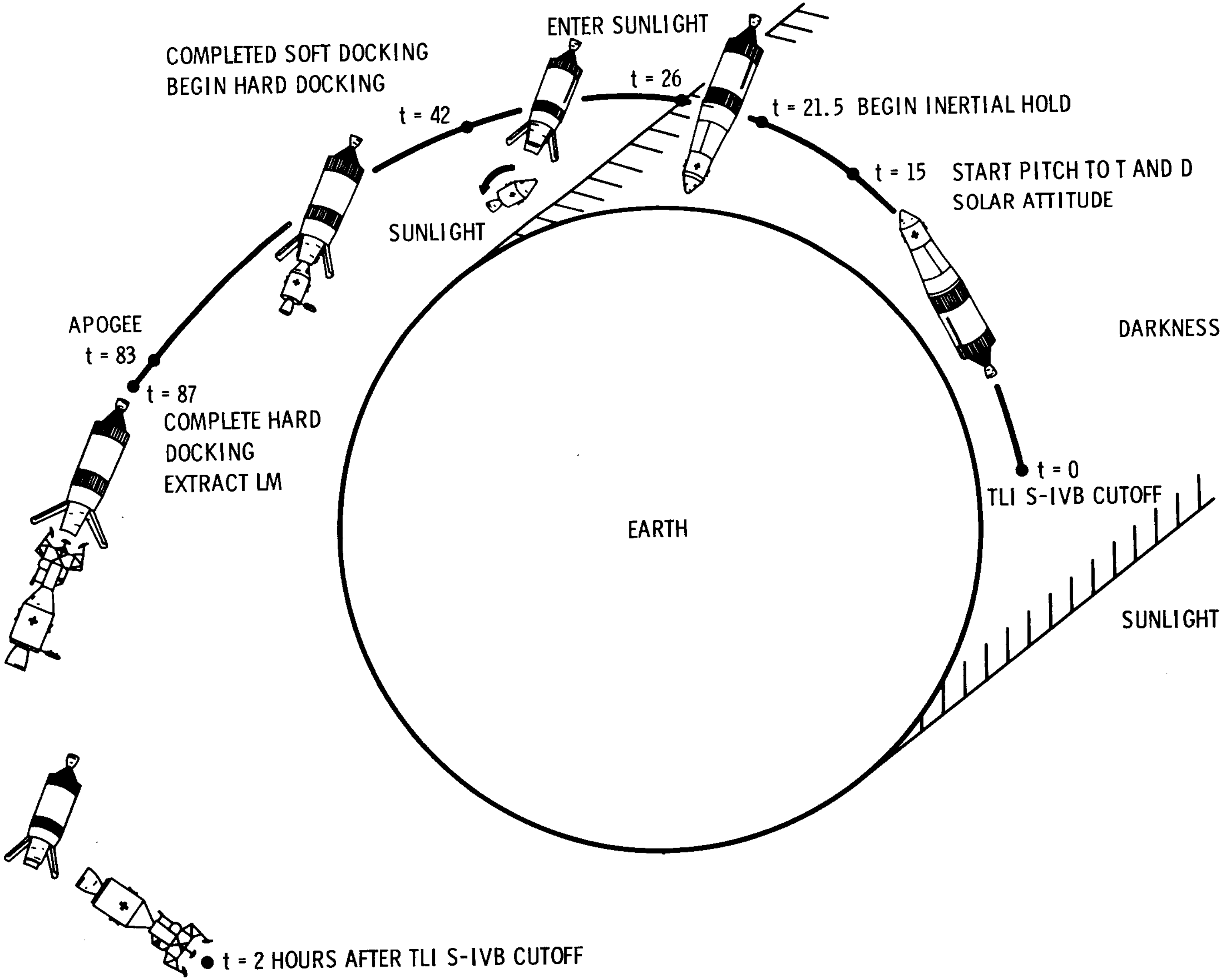
31. DEORBIT BURN

32. CM/SM SEPARATION

33. RE-ENTRY

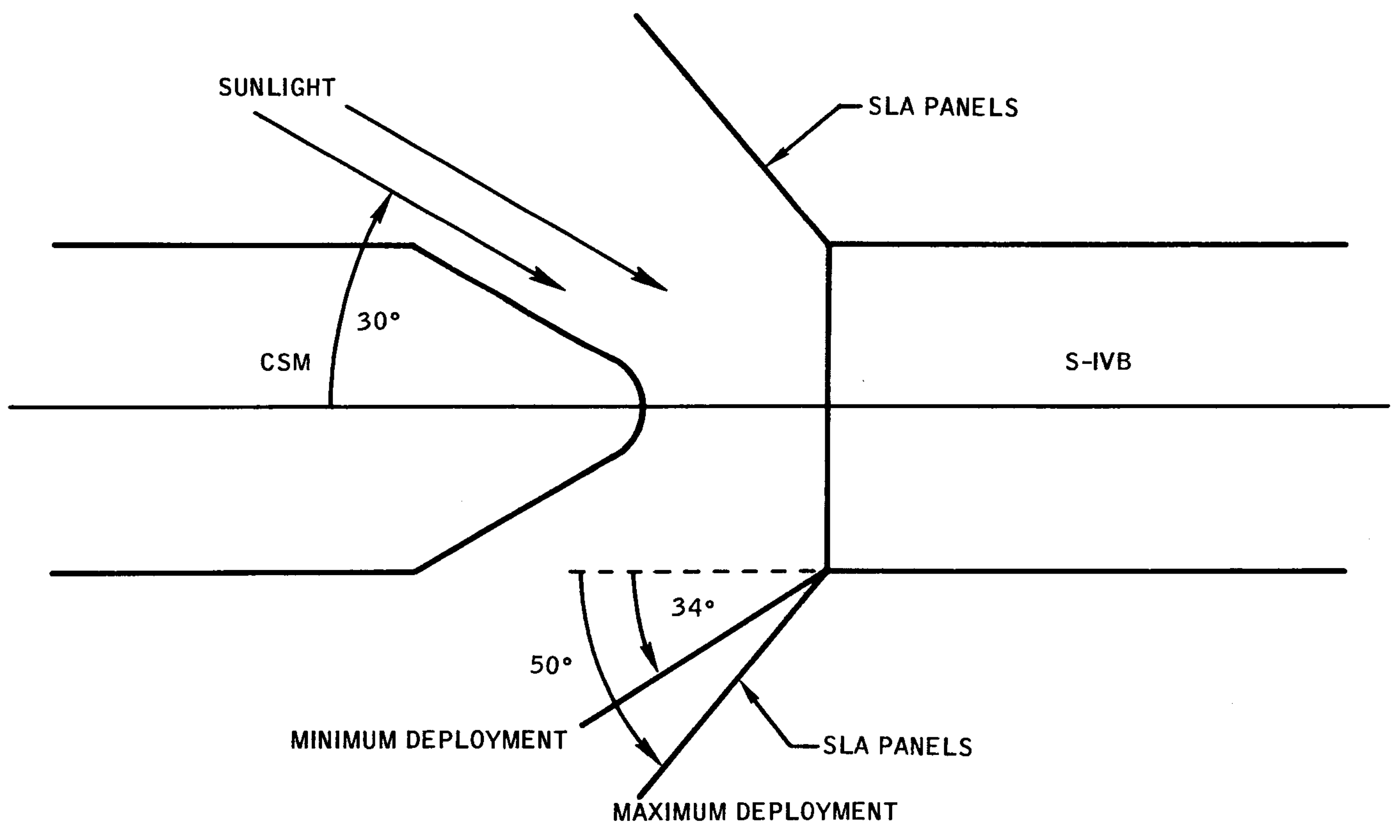
34. LANDING

Figure 6.- Diagram of events 31 through 34, fifth period of activity.



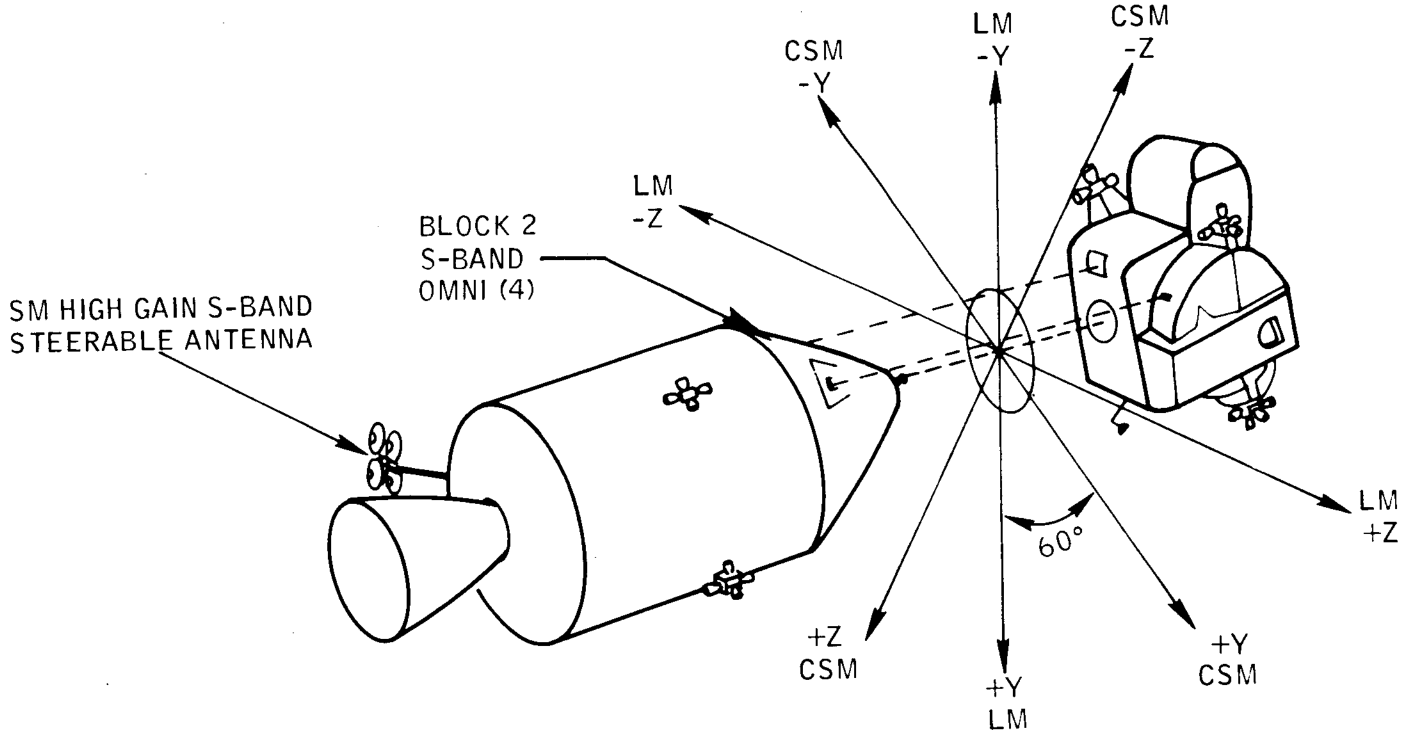
(a) Event geometry.

Figure 7. - Transposition and docking maneuver.



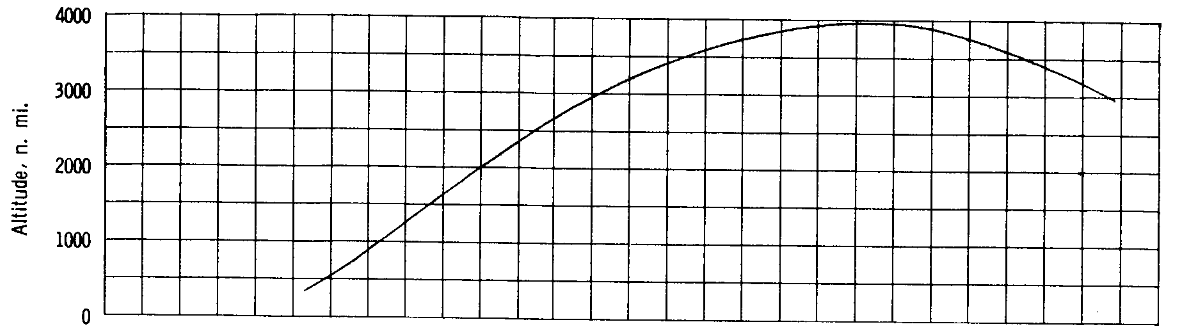
(b) Lighting geometry.

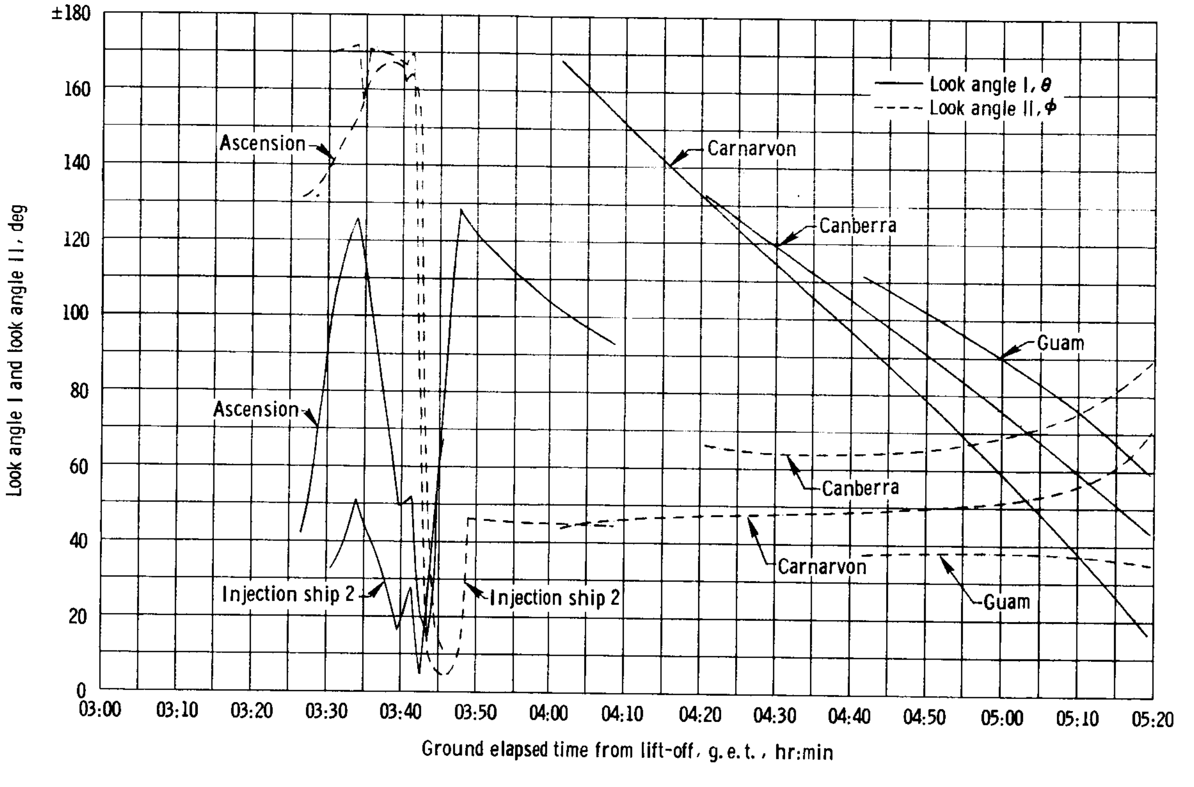
Figure 7.- Continued.



(c) Relative body axis CSM/LM.

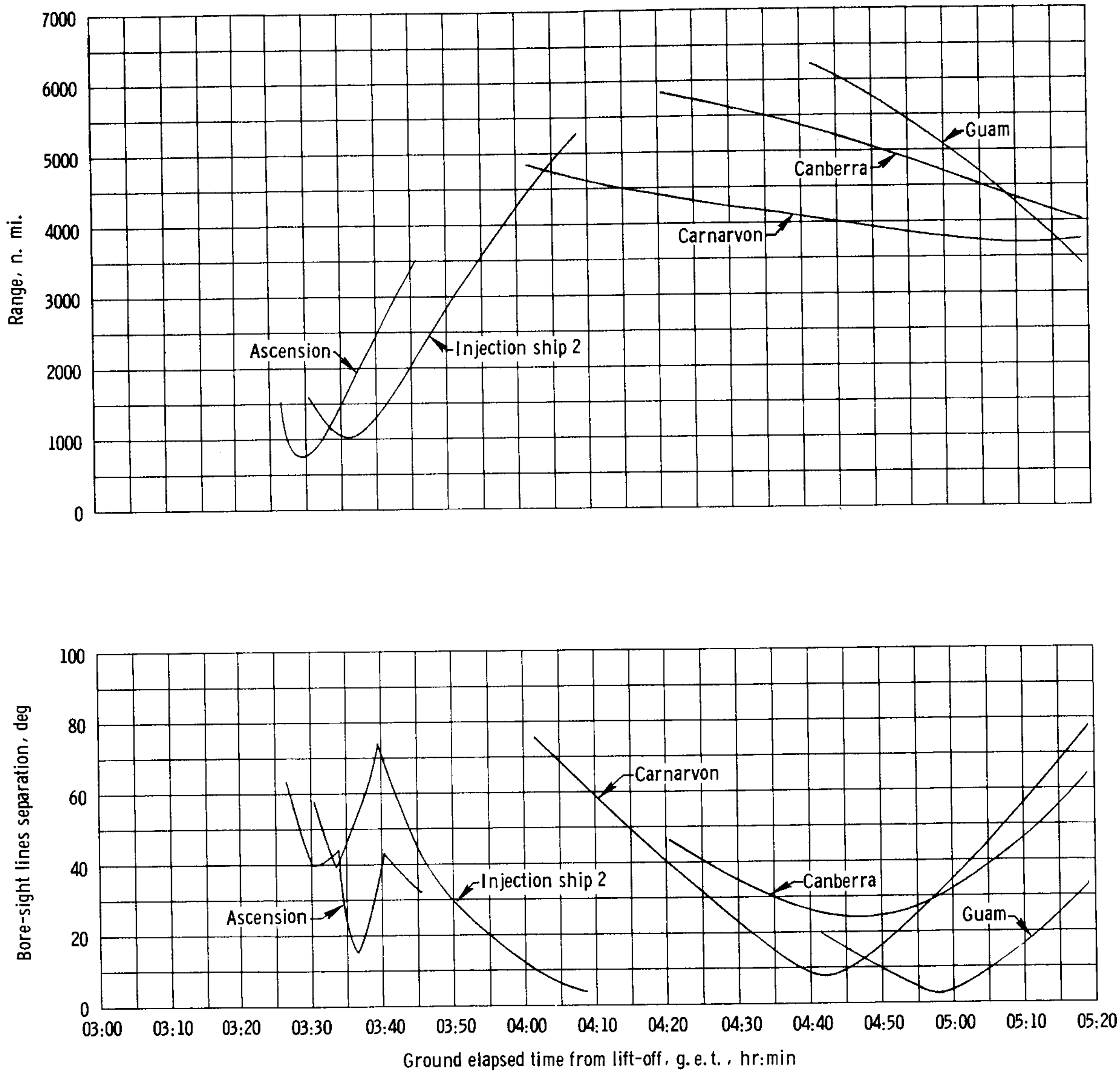
Figure 7.- Continued.





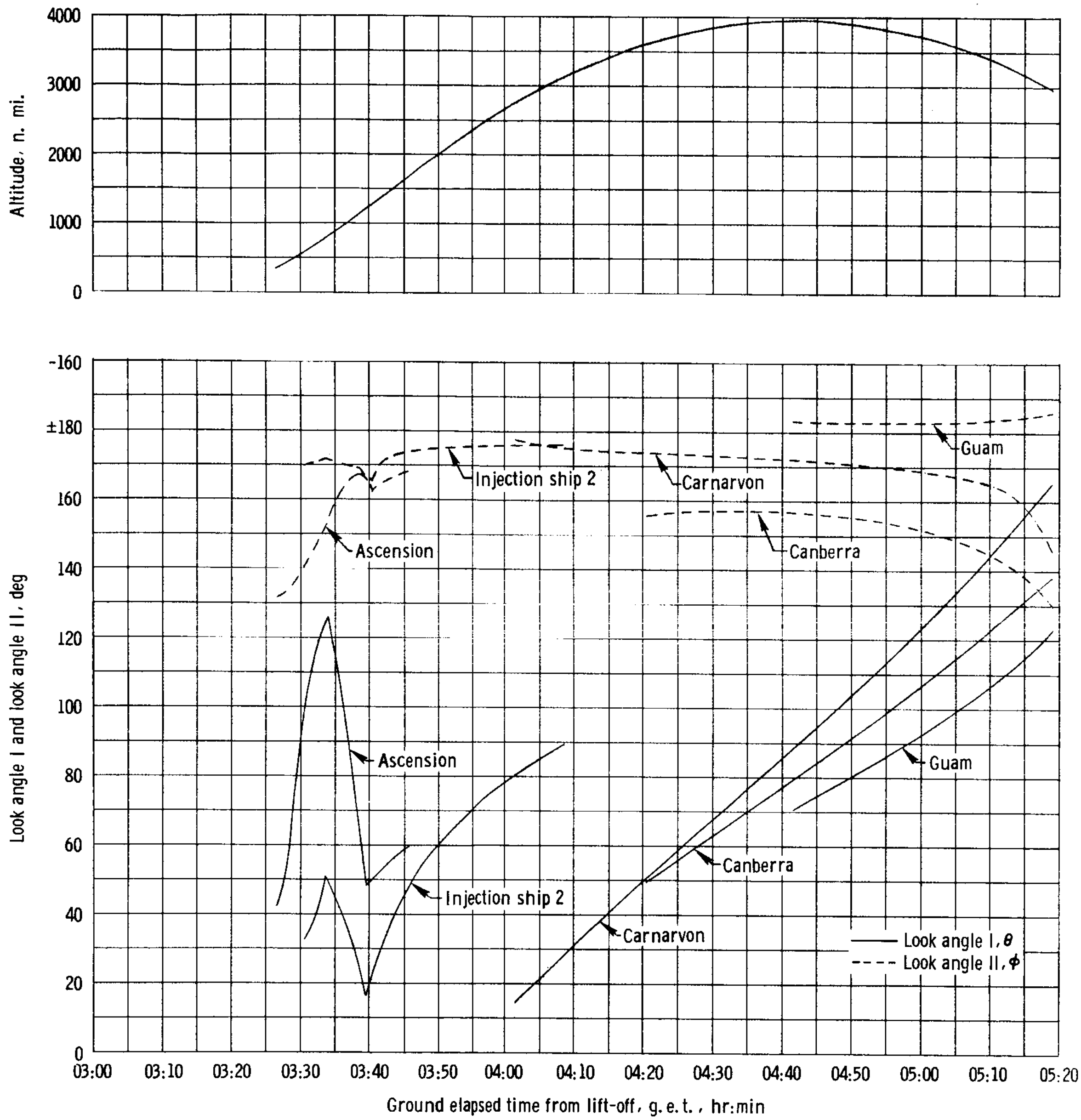
(d) CSM to ground station omni interface.

Figure 7. - Continued.



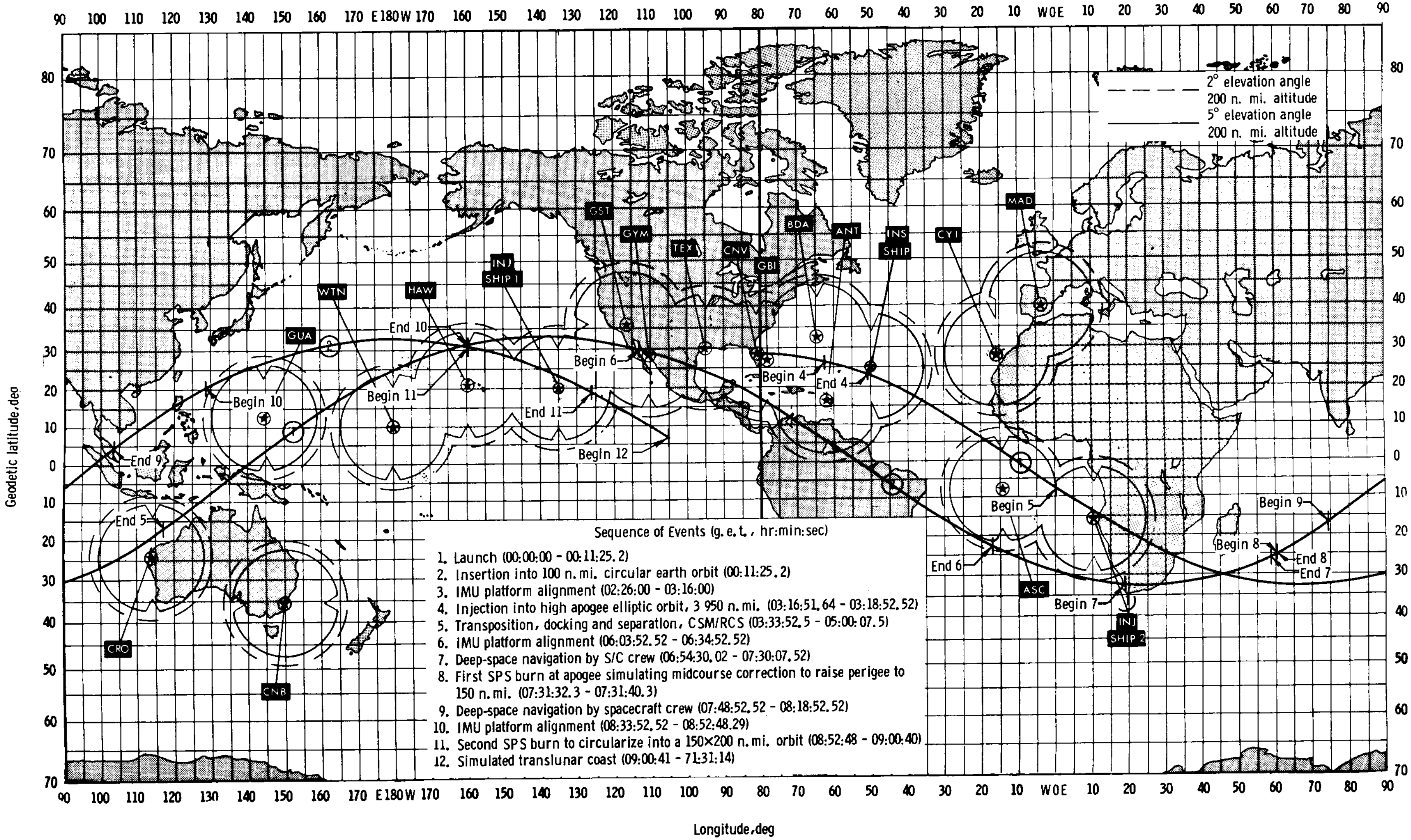
(e) S-IVB to ground station directional antenna interface.

Figure 7. - Continued.



(f) S-IVB to ground station omni interface.

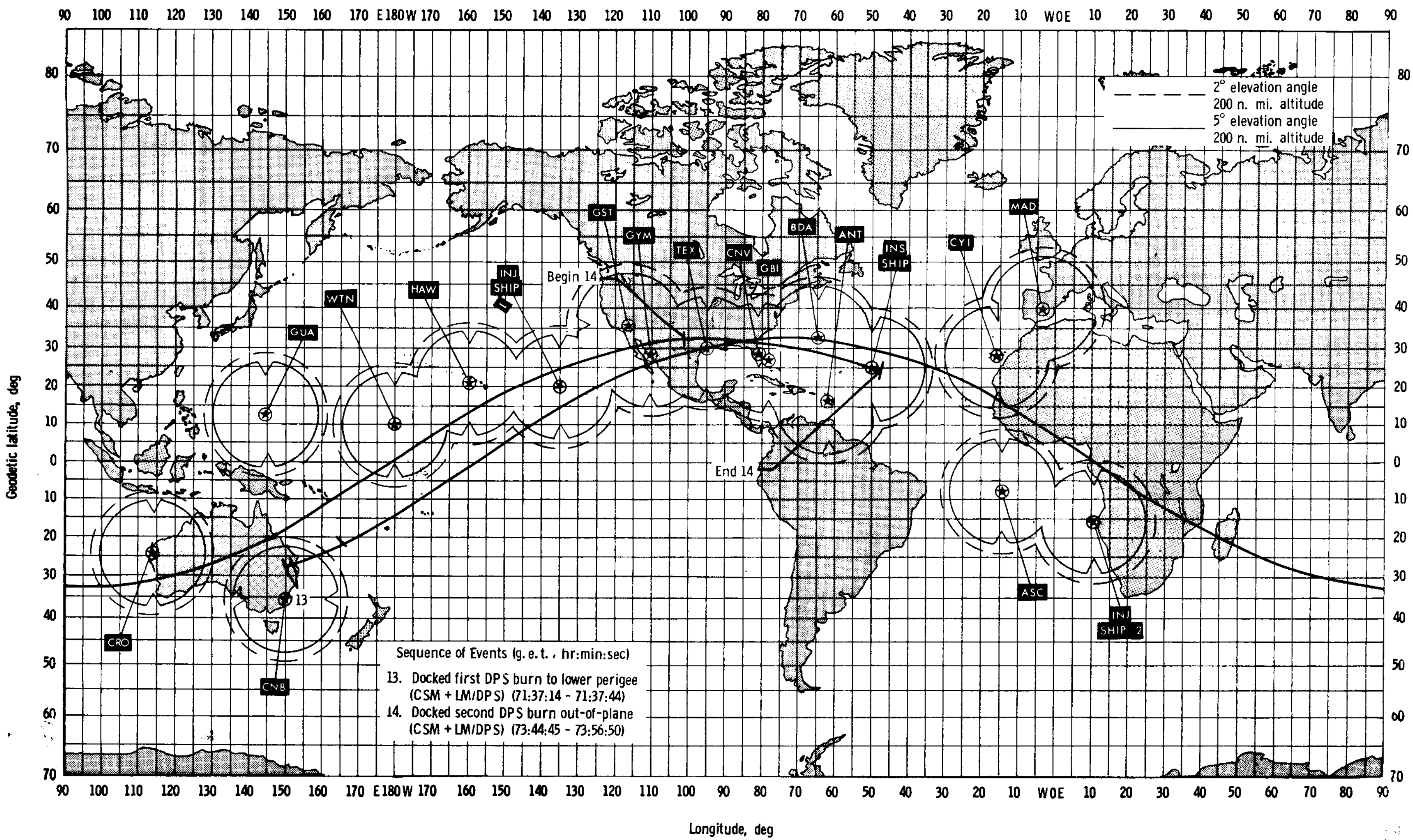
Figure 7. - Concluded.



(a) First period of activity.

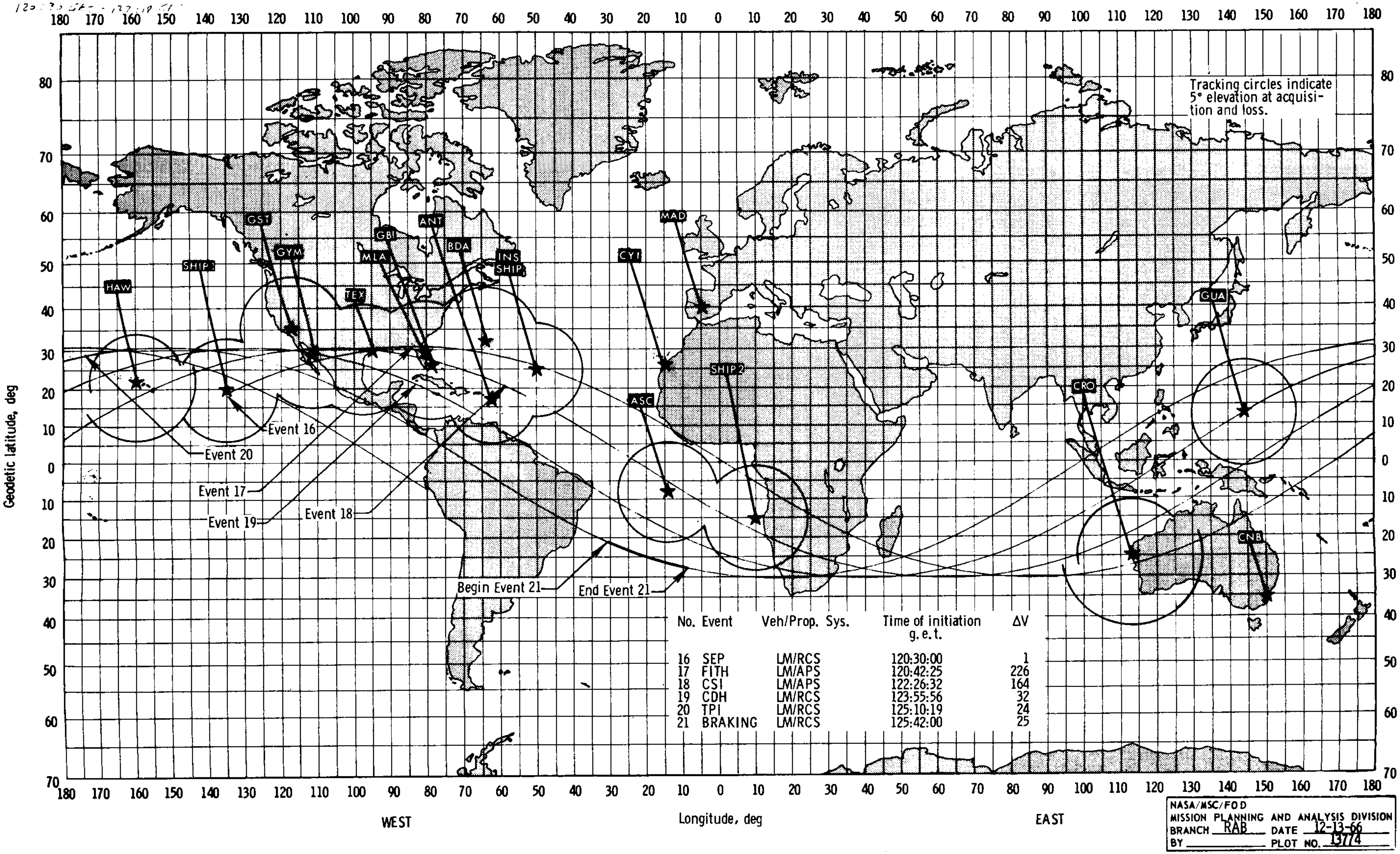
Figure 8. - AS-503A ground tracks (RT).

APOLLO TRACKING NETWORK
(Land based unified S-band radar)



(b) Second period of activity.

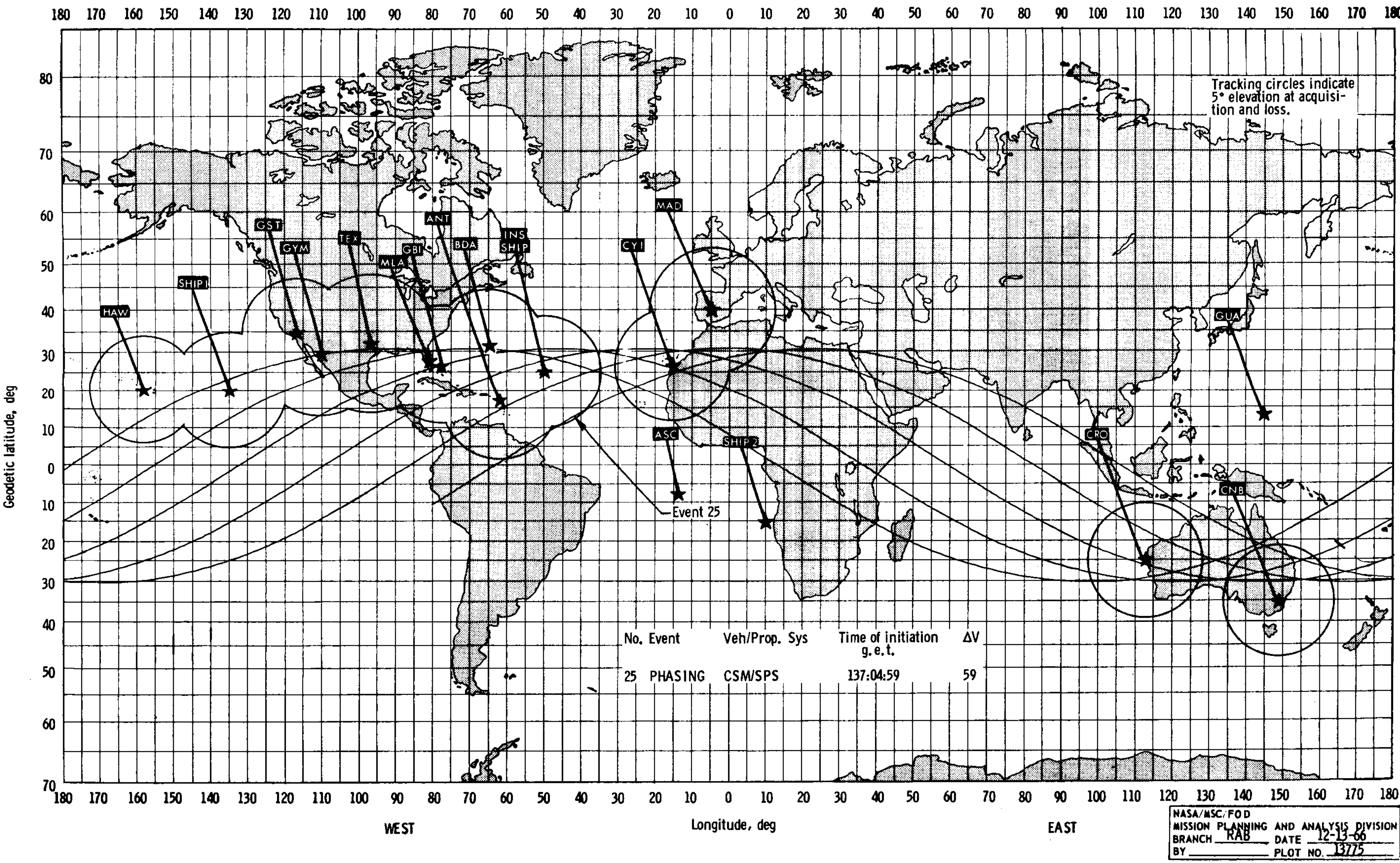
Figure 8. - Continued.



No.	Event	Veh/Prop. Sys.	Time of initiation g.e.t.	ΔV
16	SEP	LM/RCS	120:30:00	1
17	FITH	LM/APS	120:42:25	226
18	CSI	LM/APS	122:26:32	164
19	CDH	LM/RCS	123:55:56	32
20	TPI	LM/RCS	125:10:19	24
21	BRAKING	LM/RCS	125:42:00	25

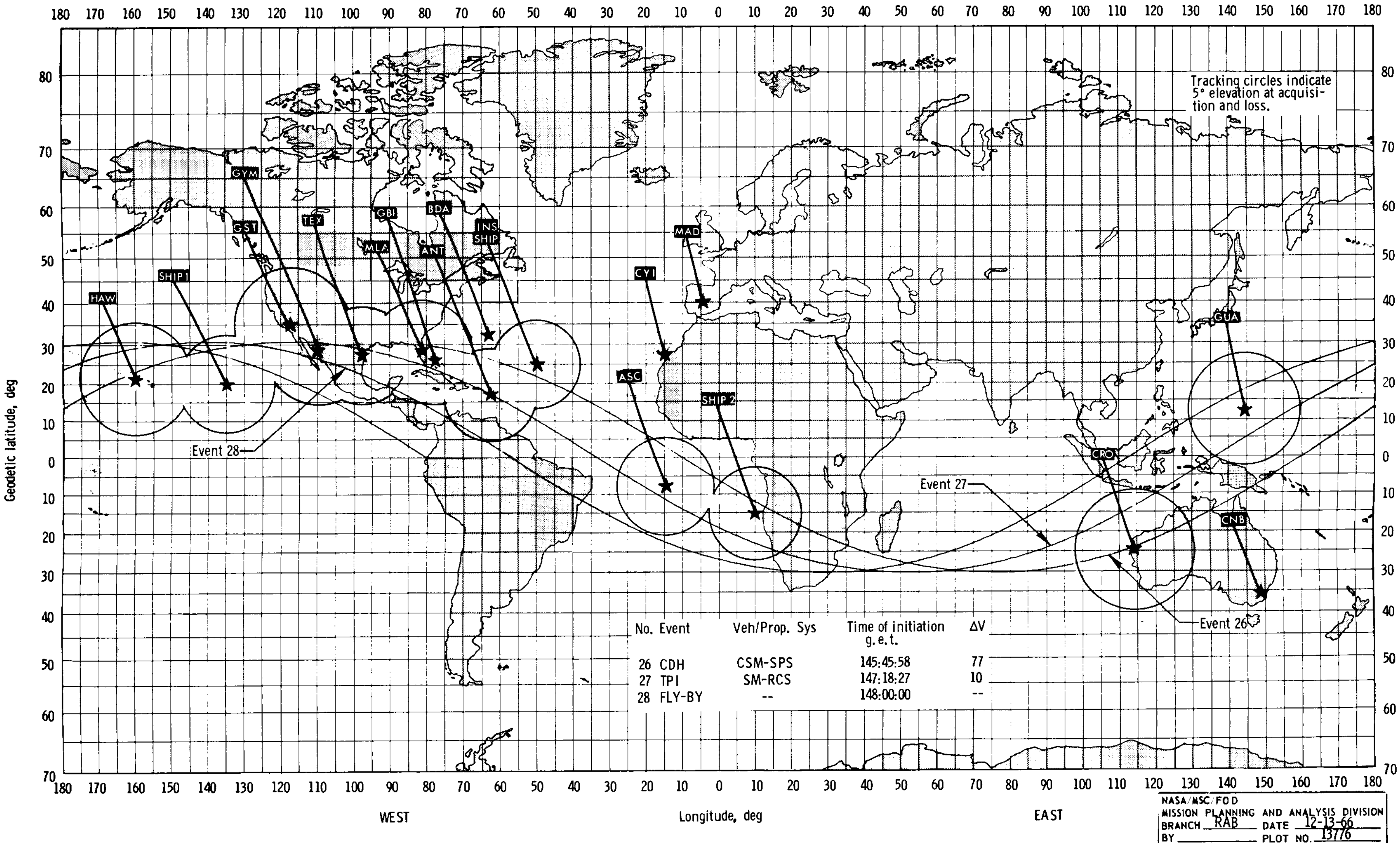
(c) Third period of activity, (LM) separation through rendezvous.

Figure 8. - Continued.



(d) Fourth period of activity, (CSM) event 25.

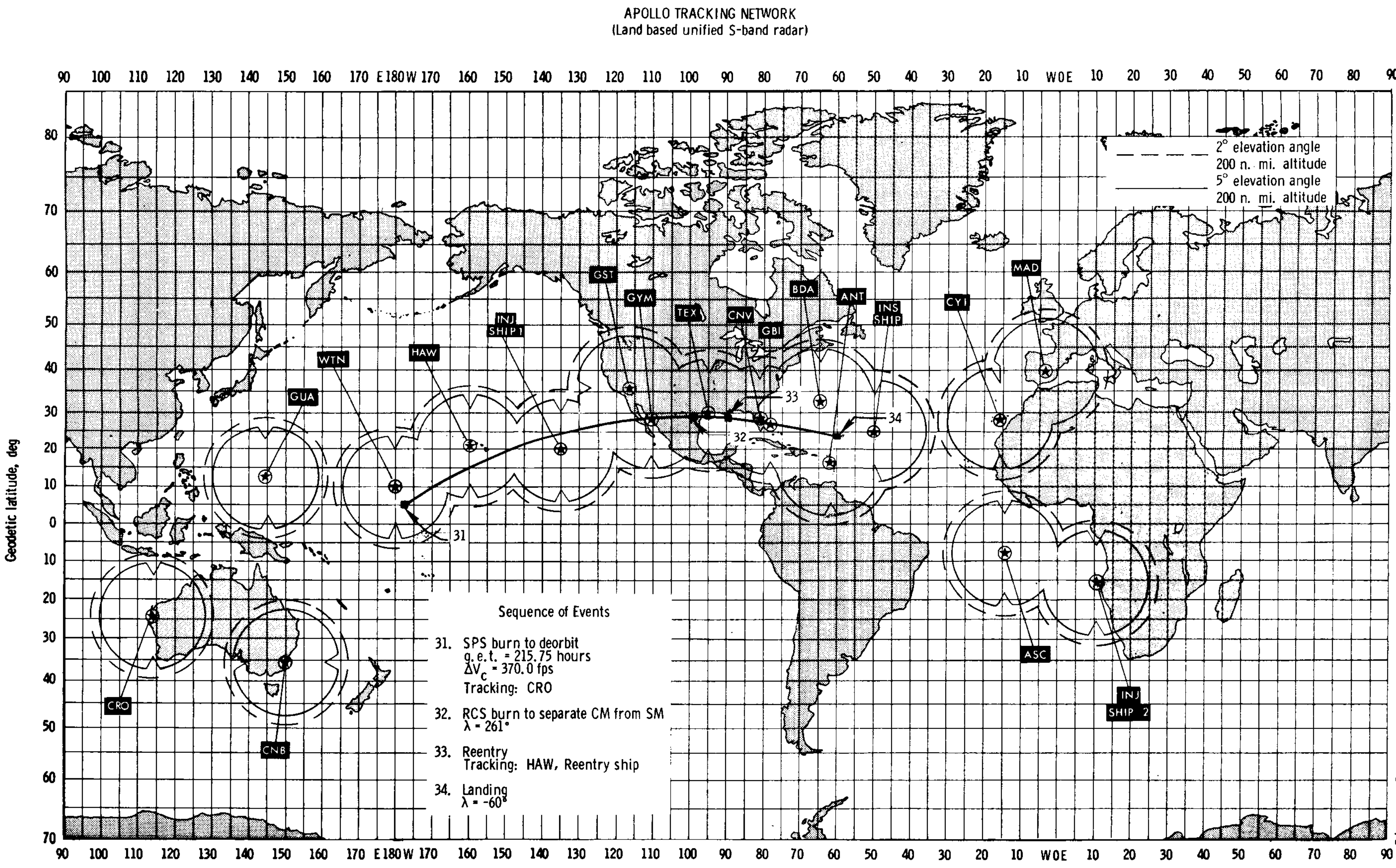
Figure 8. - Continued.



No.	Event	Veh/Prop. Sys	Time of initiation g.e.t.	ΔV
26	CDH	CSM-SPS	145:45:58	77
27	TPI	SM-RCS	147:18:27	10
28	FLY-BY	--	148:00:00	--

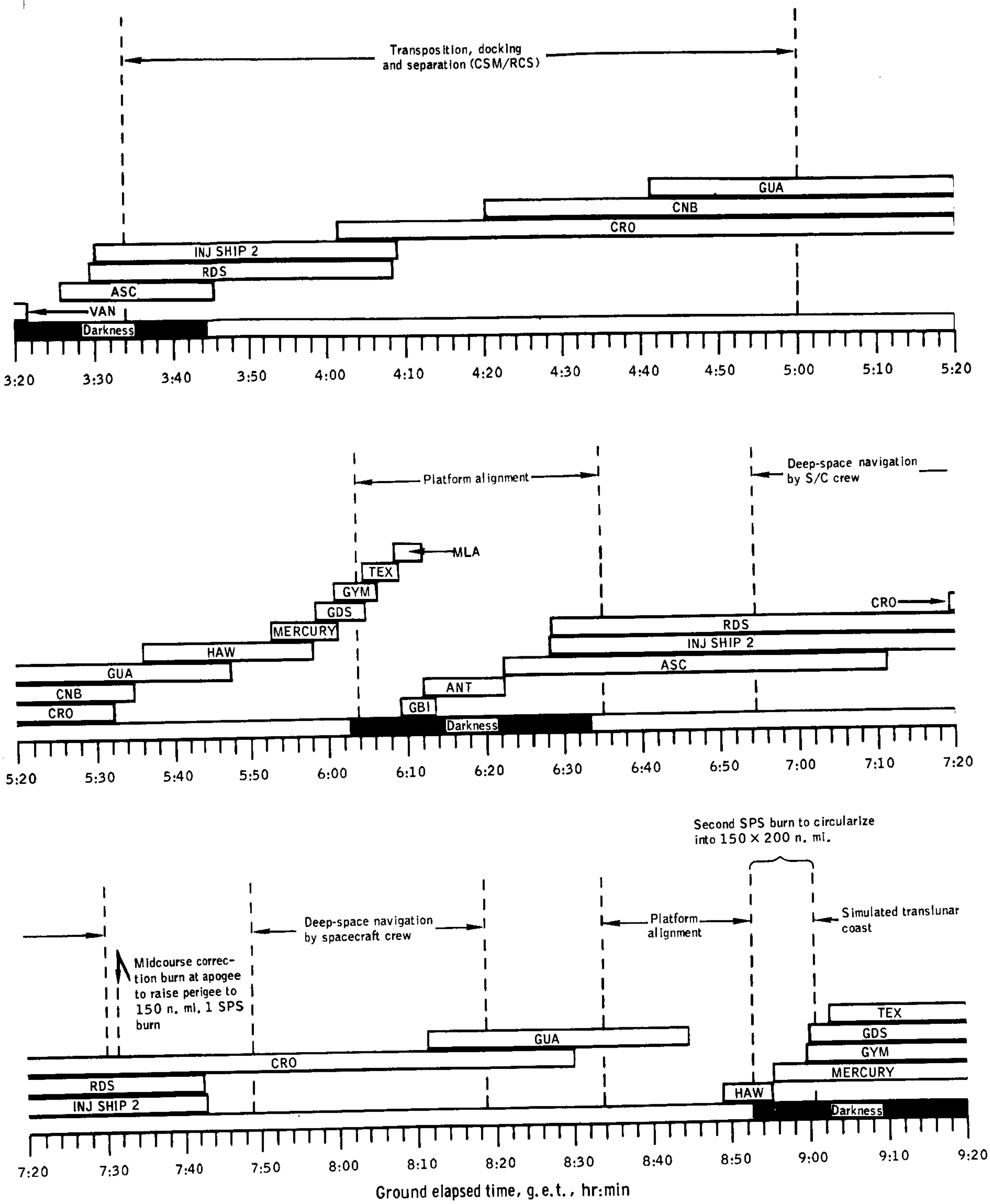
(e) Fourth period of activity, events 26 through 28.

Figure 8. - Continued.



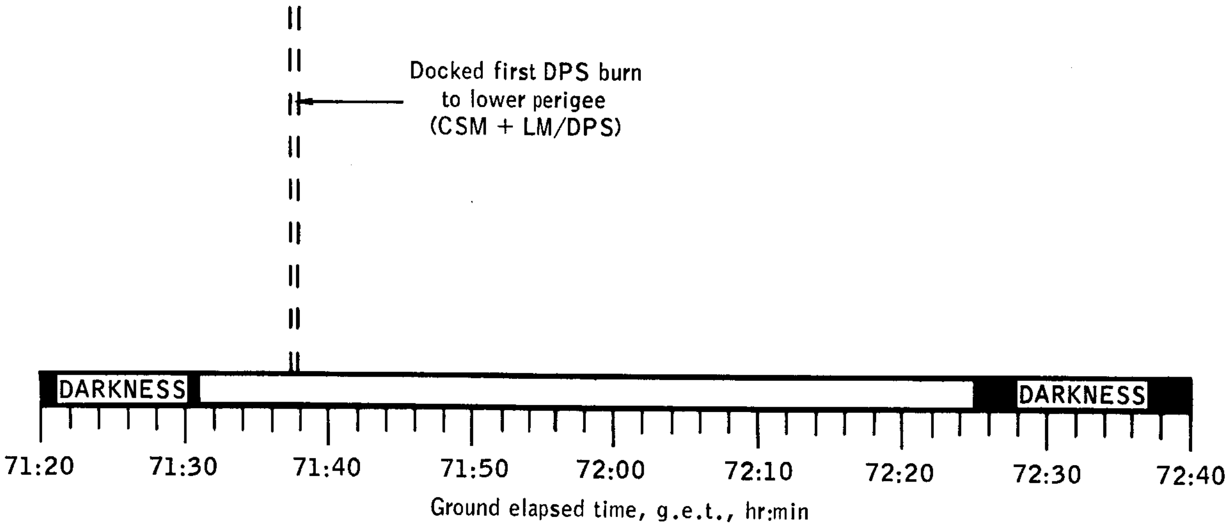
(f) Fifth period of activity.

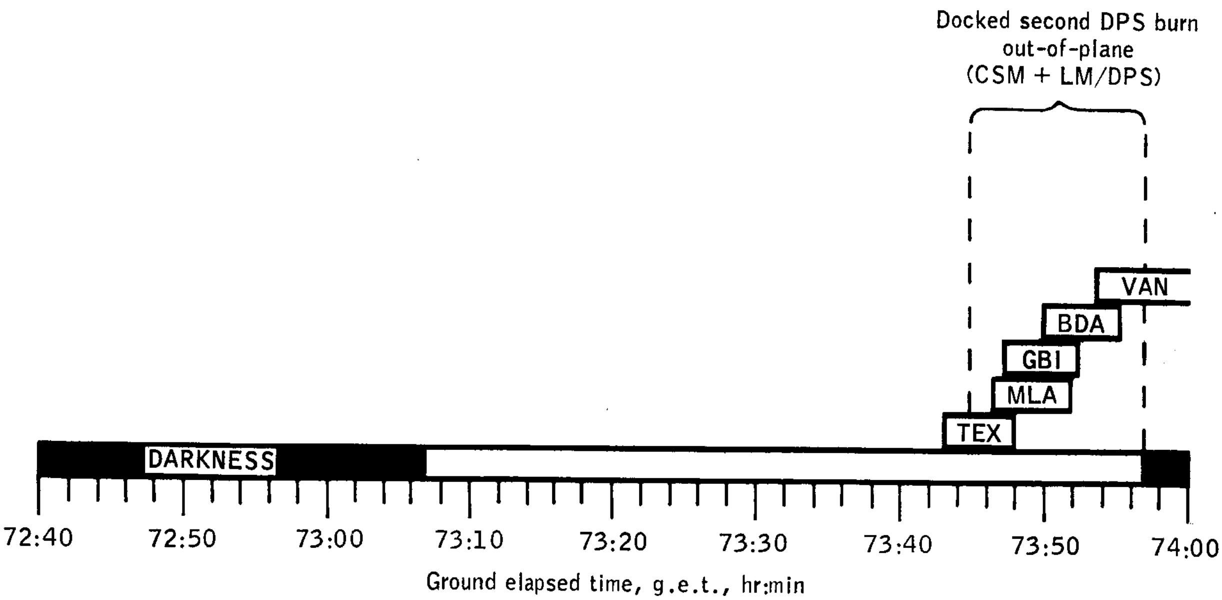
Figure 8. - Concluded.



(a) First period of activity.

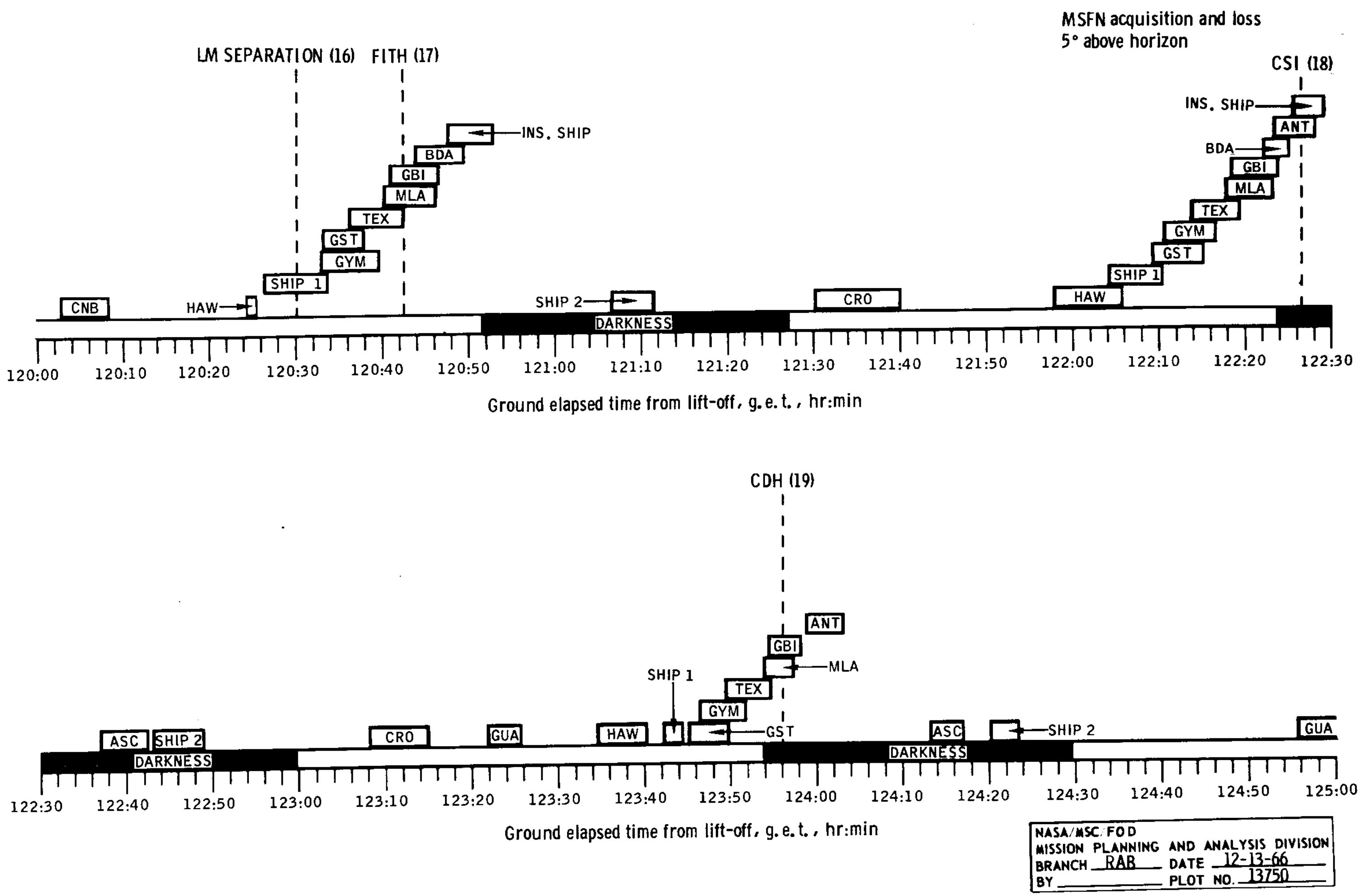
Figure 9. - AS-503A MSFN USB station viewing, major events, daylight-darkness timeline bar chart.





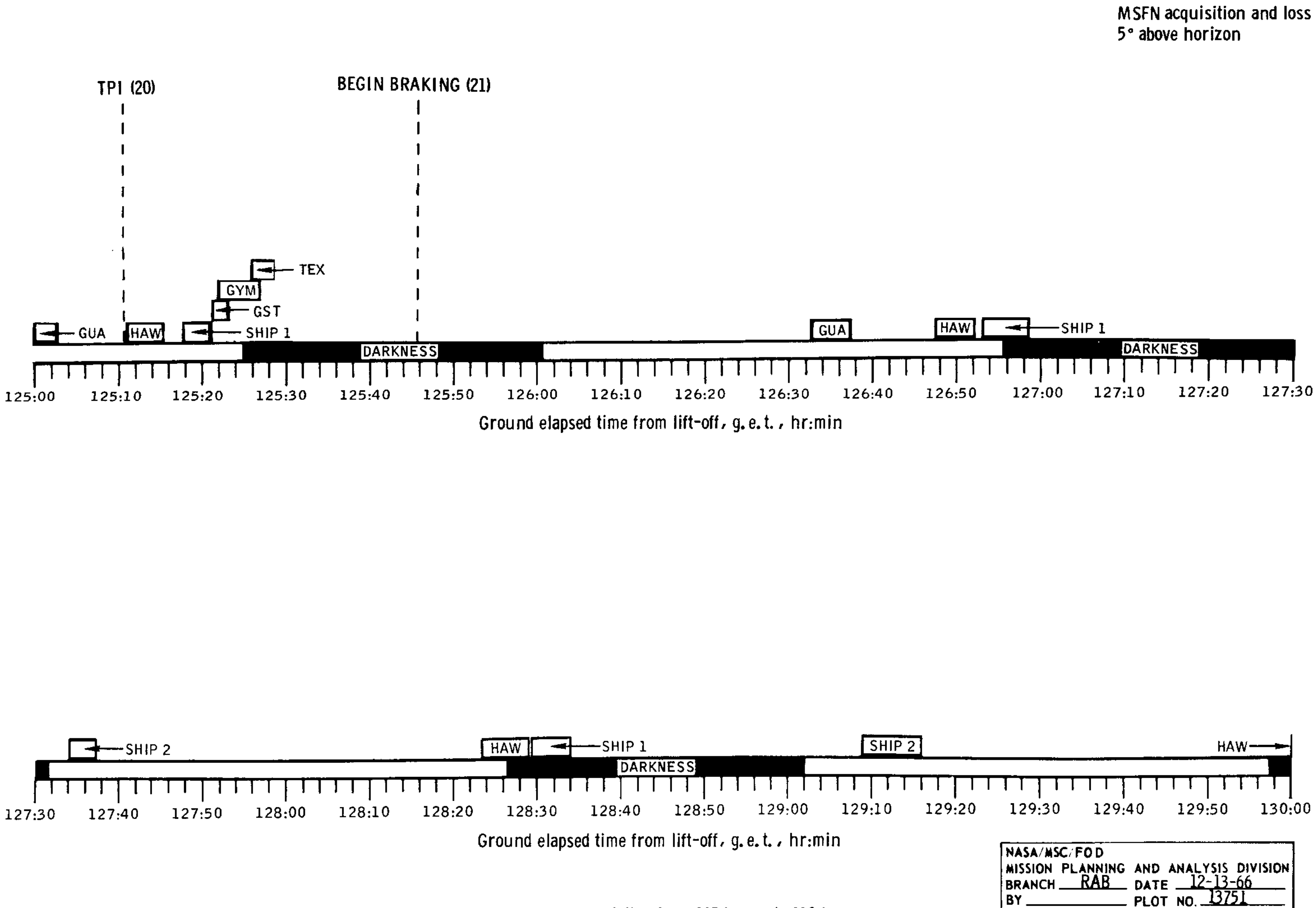
(b) Second period of activity.

Figure 9. - Continued.



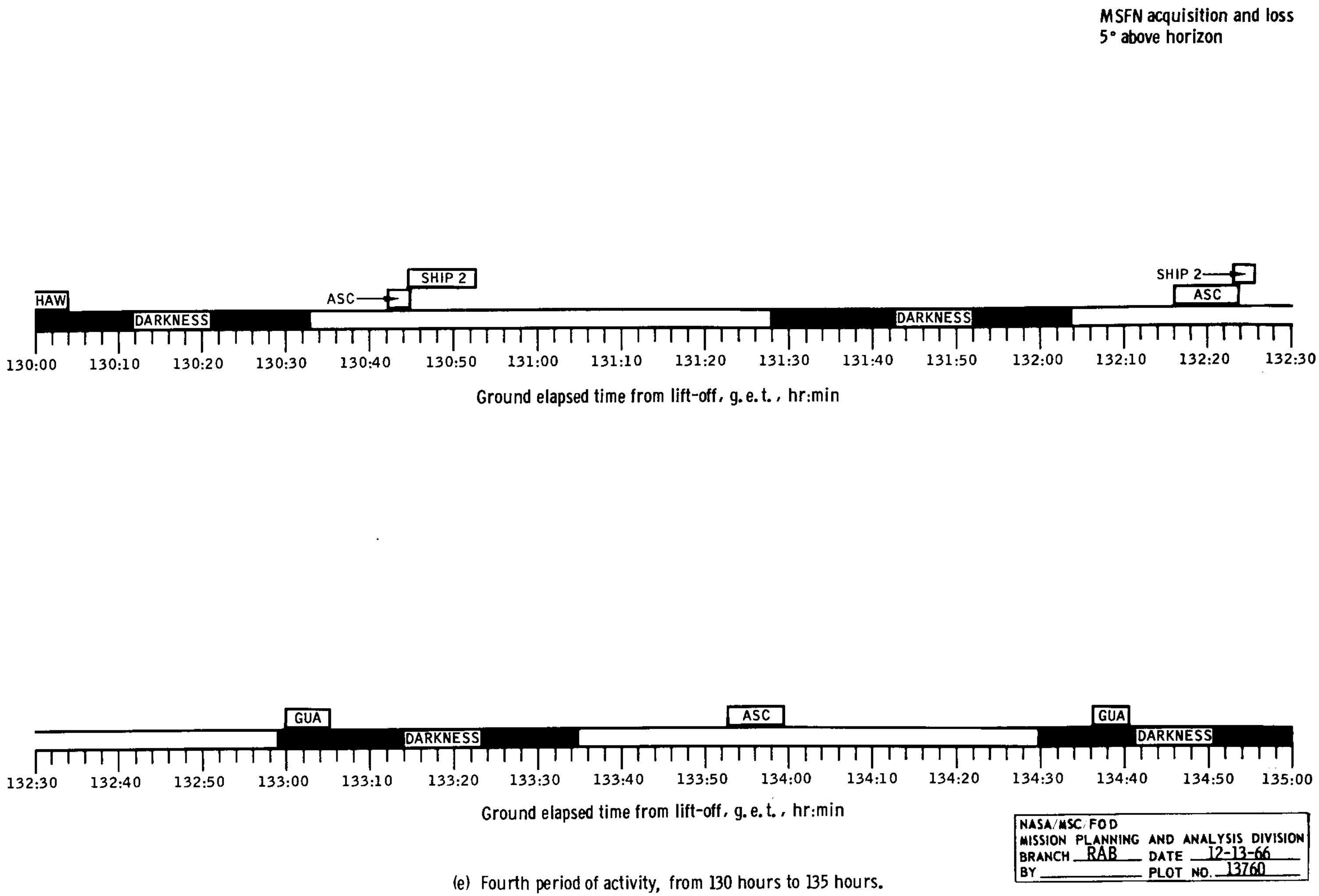
(c) Third period of activity, from 120 hours to 125 hours.

Figure 9. - Continued.



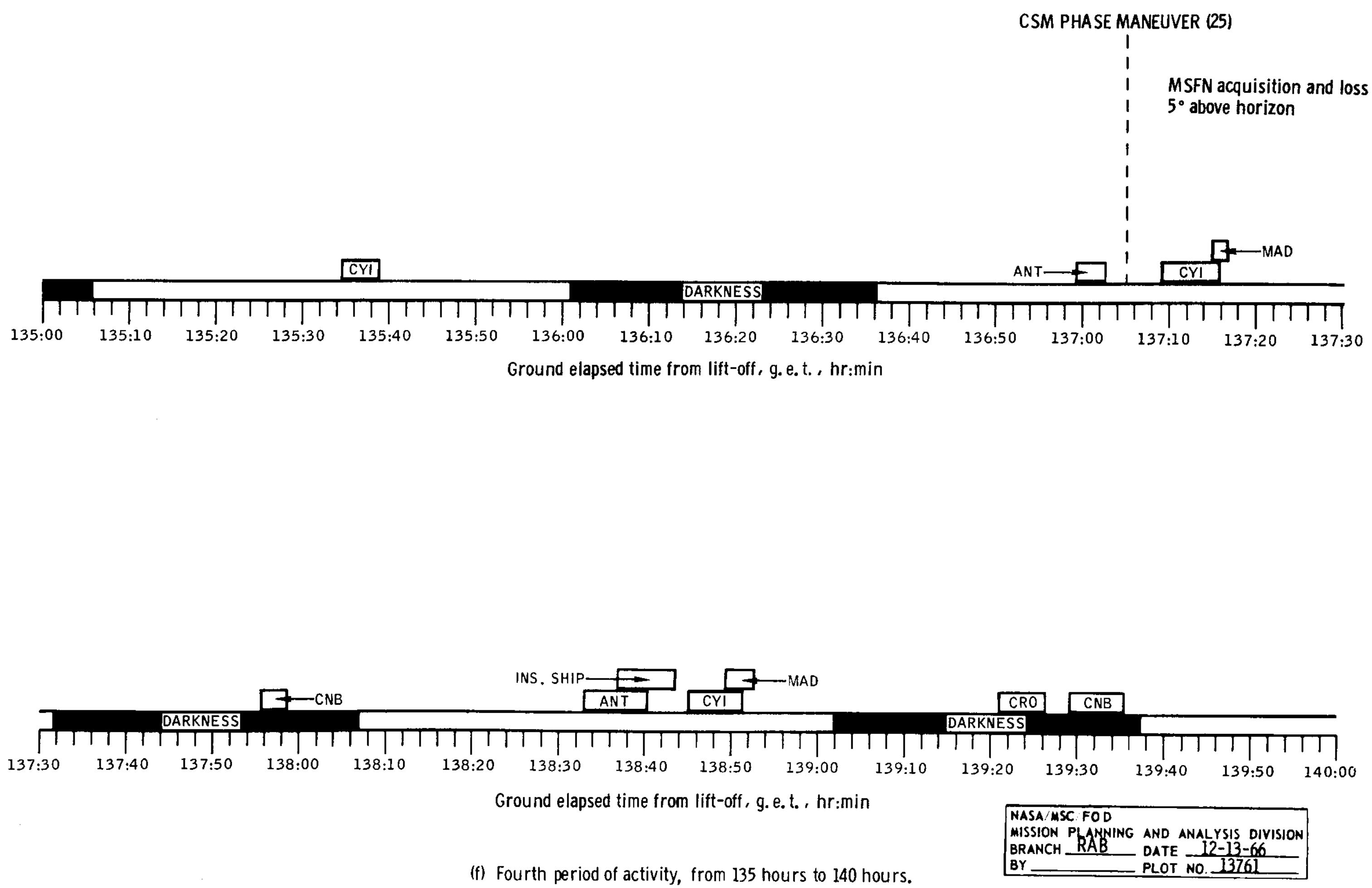
(d) Third period of activity, from 125 hours to 130 hours.

Figure 9. - Continued.



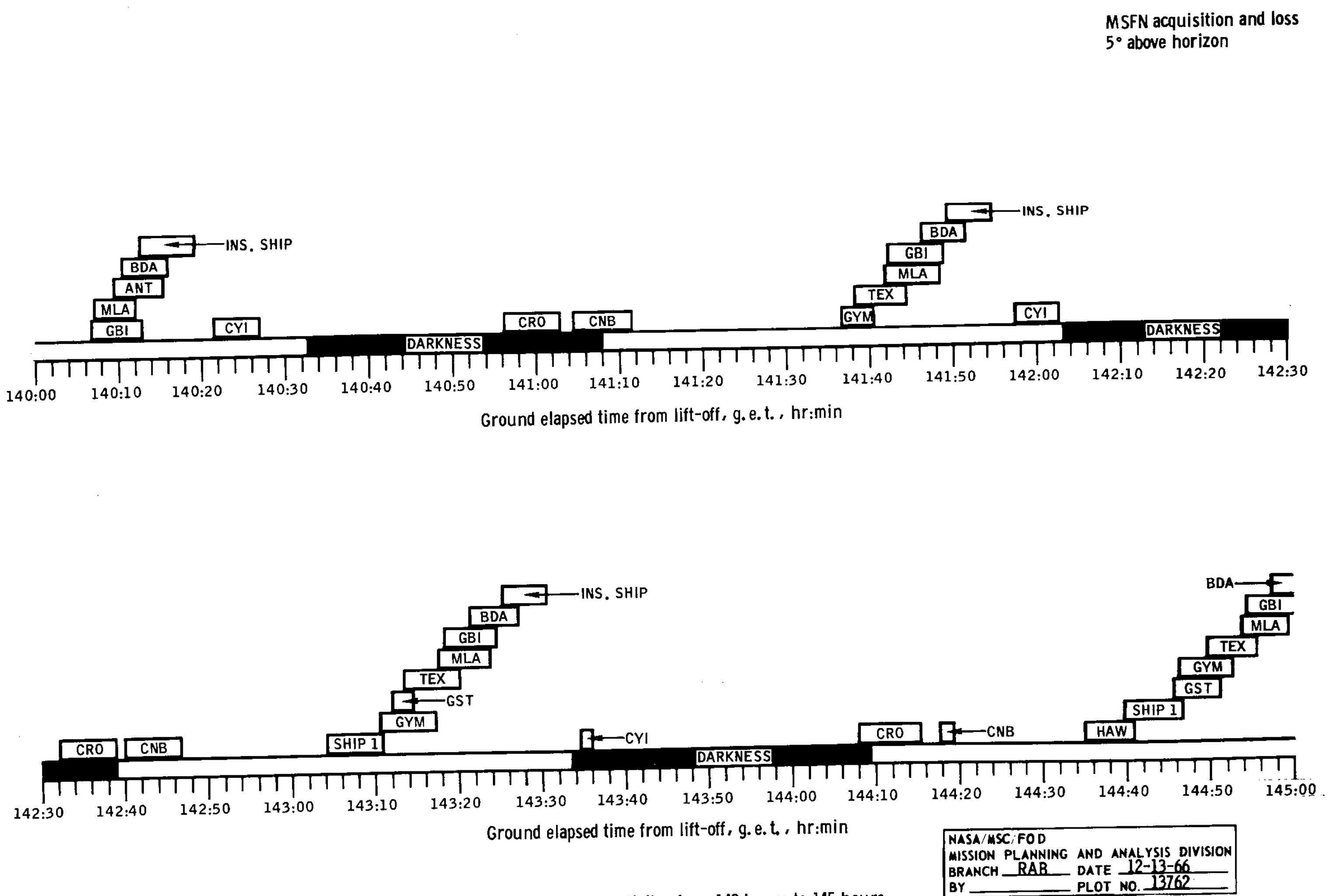
(e) Fourth period of activity, from 130 hours to 135 hours.

Figure 9. - Continued.



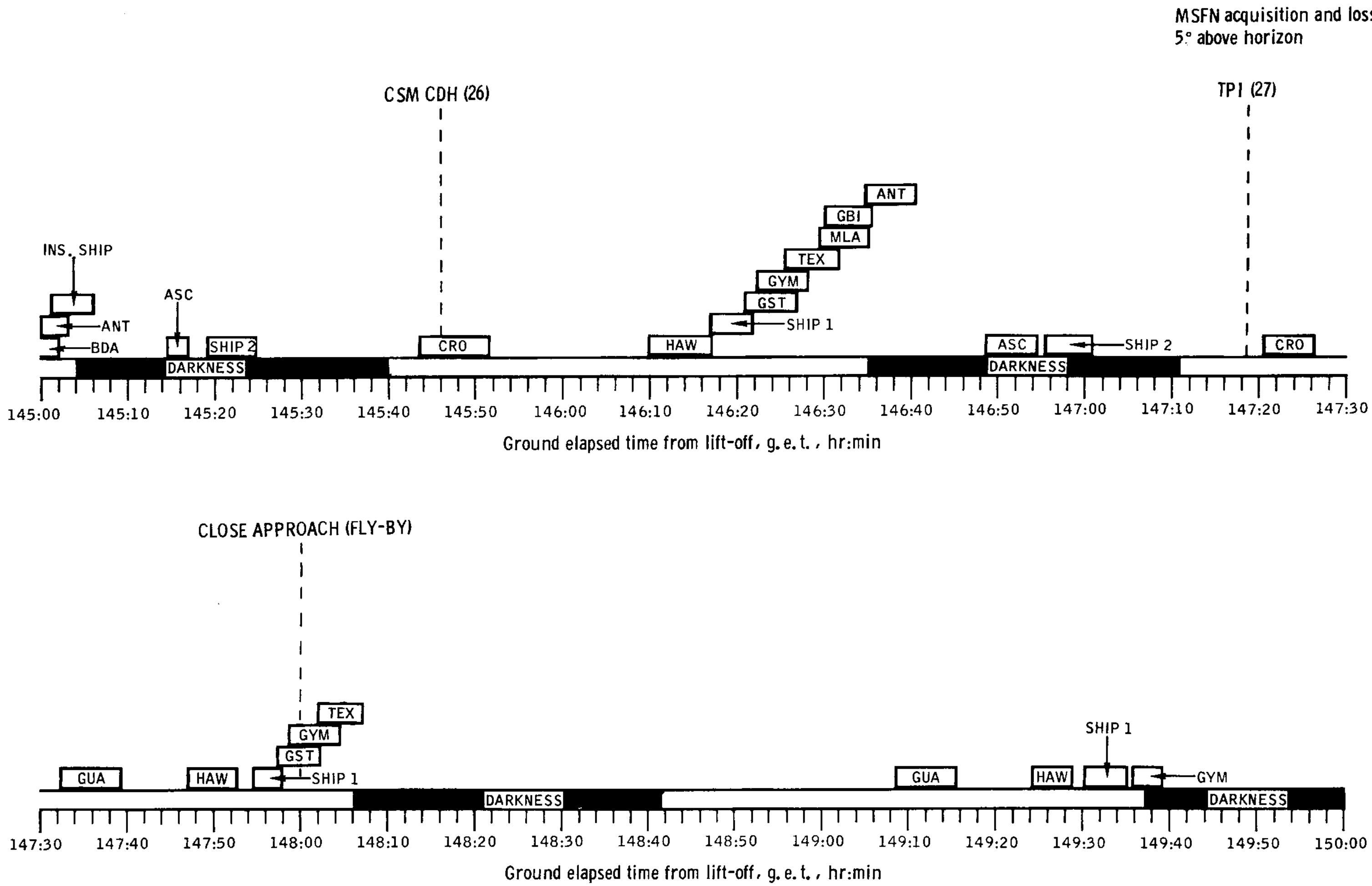
(f) Fourth period of activity, from 135 hours to 140 hours.

Figure 9. - Continued.



(g) Fourth period of activity, from 140 hours to 145 hours.

Figure 9. - Continued.



(h) Fourth period of activity, from 145 hours to 150 hours.

Figure 9. - Concluded.

NASA/MSC/FOD
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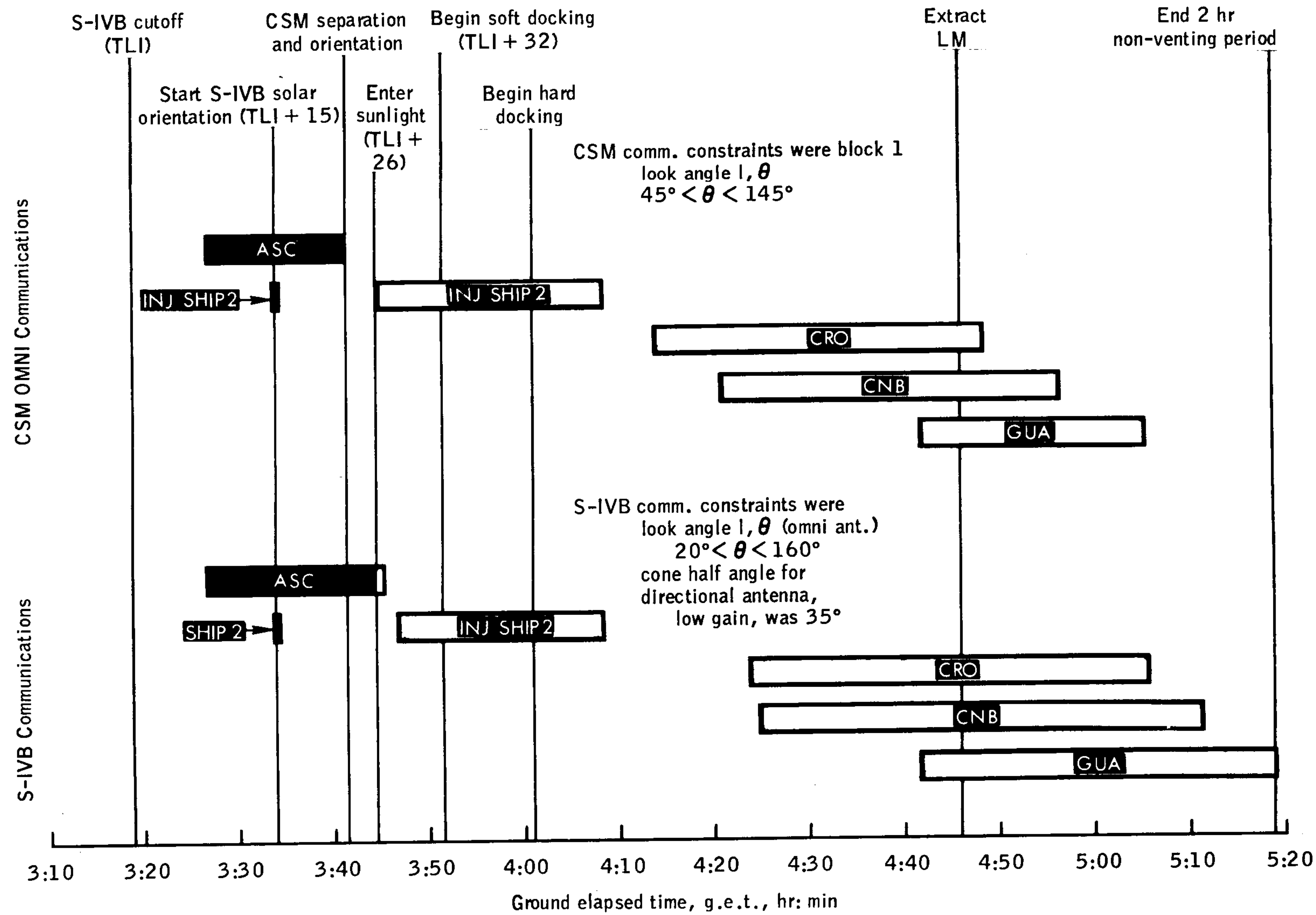
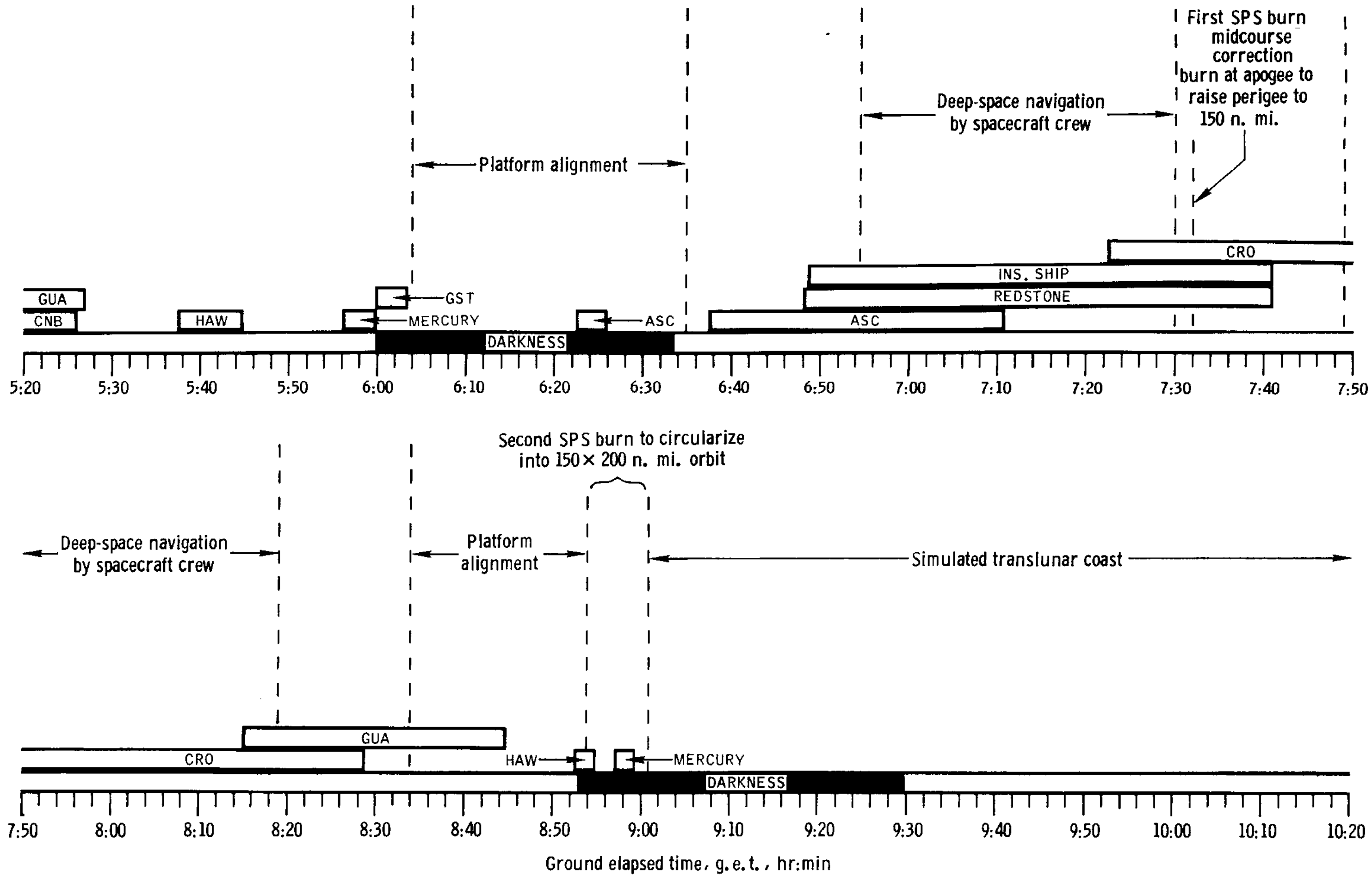
BY ______ PLOT NO. 13763



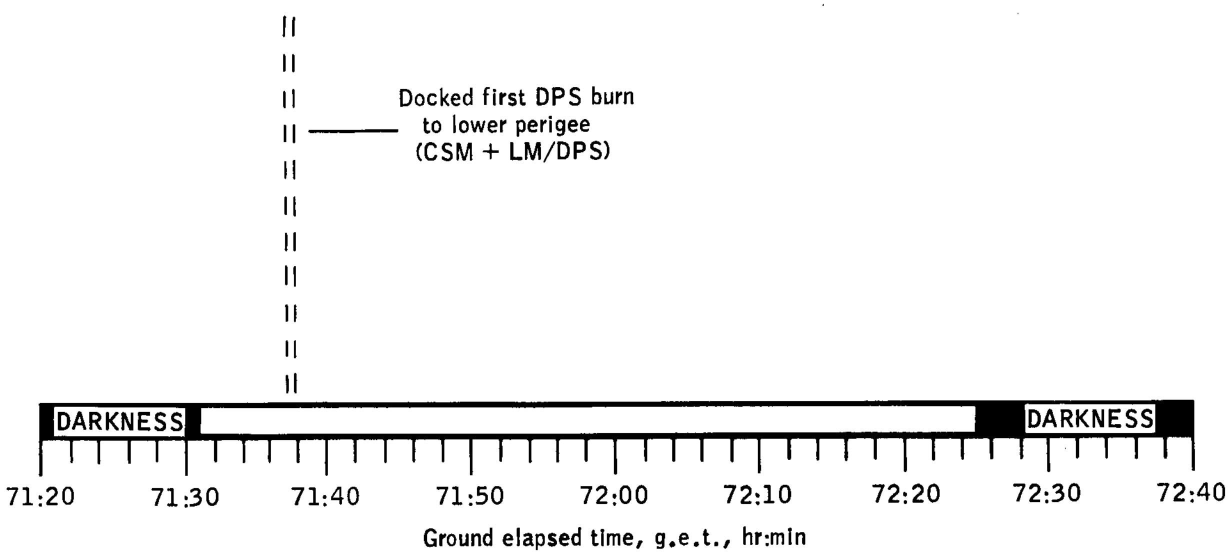
(a) First period of activity, from 3:20 hours to 5:20 hours.

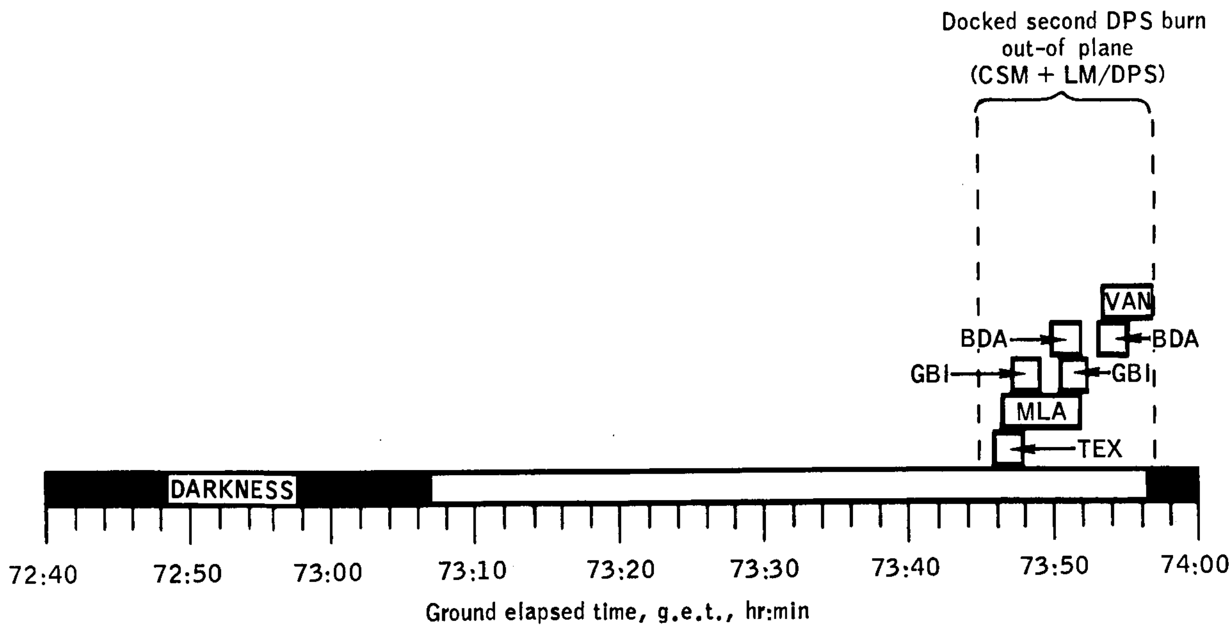
Figure 10. - USB Omni communications bar charts.



(b) First period of activity, from 5:20 hours to 10:20 hours.

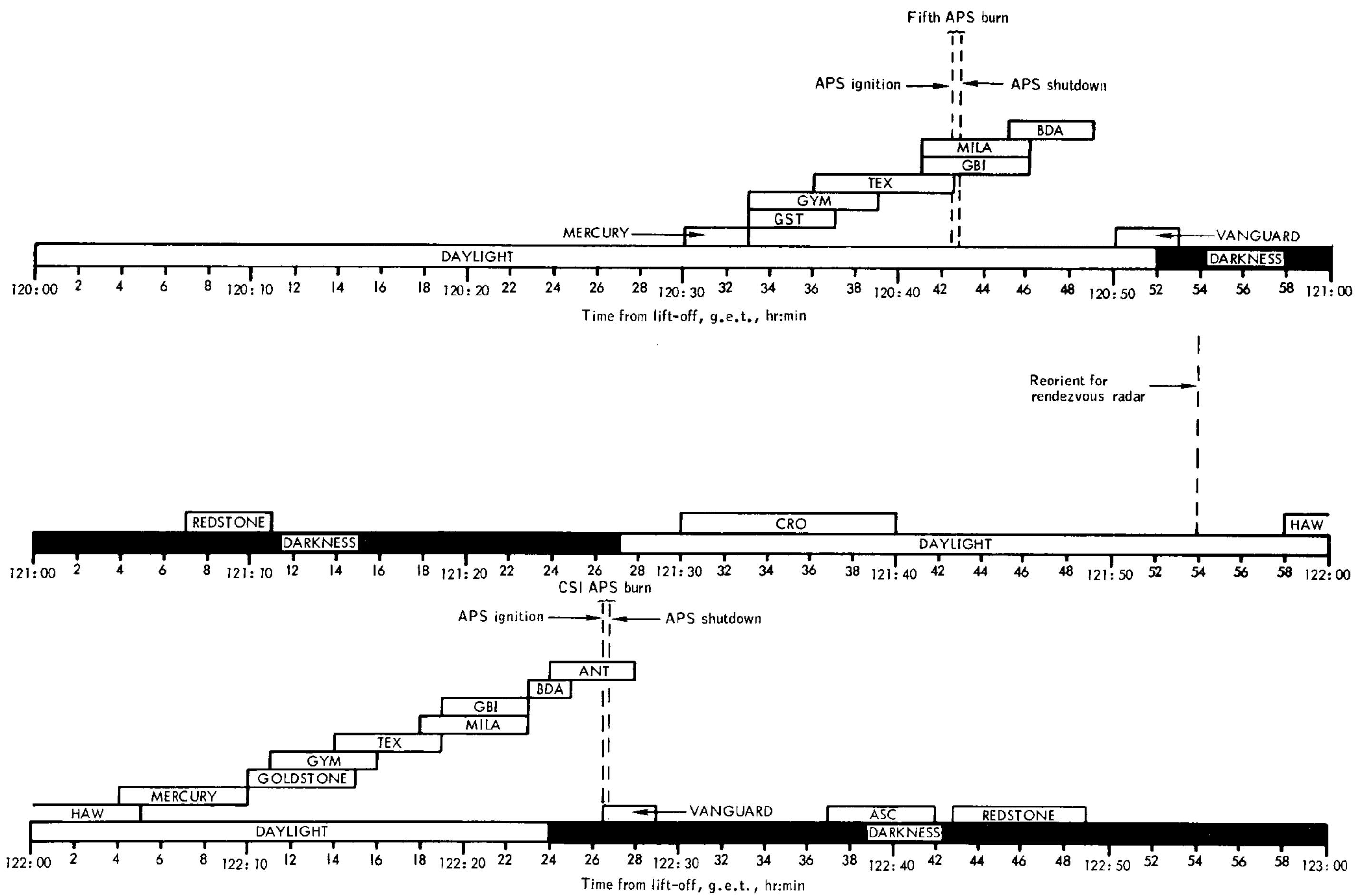
Figure 10. - Continued.





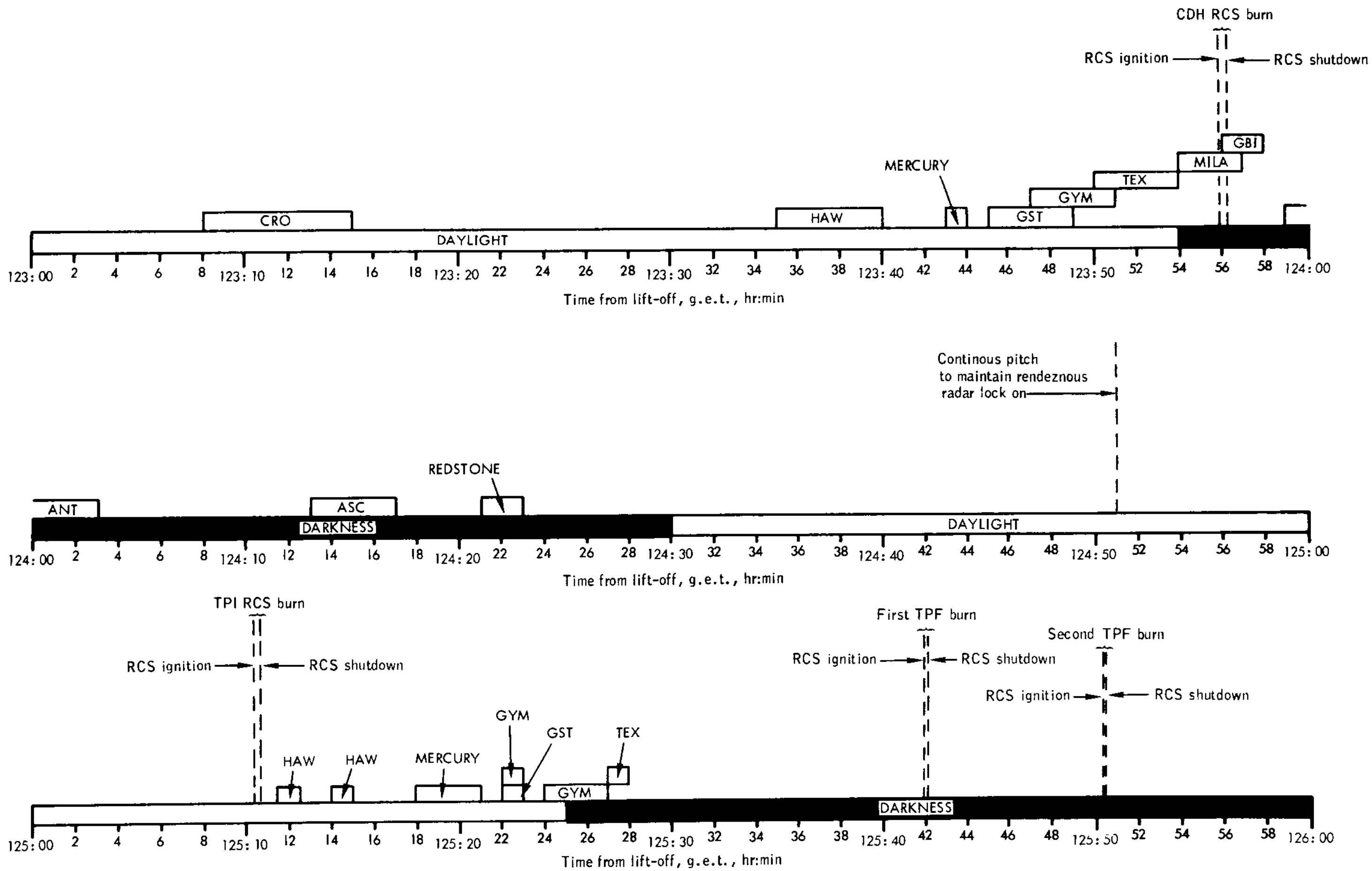
(c) Second period of activity, from 71:20 hours to 74:00 hours.

Figure 10. - Continued.



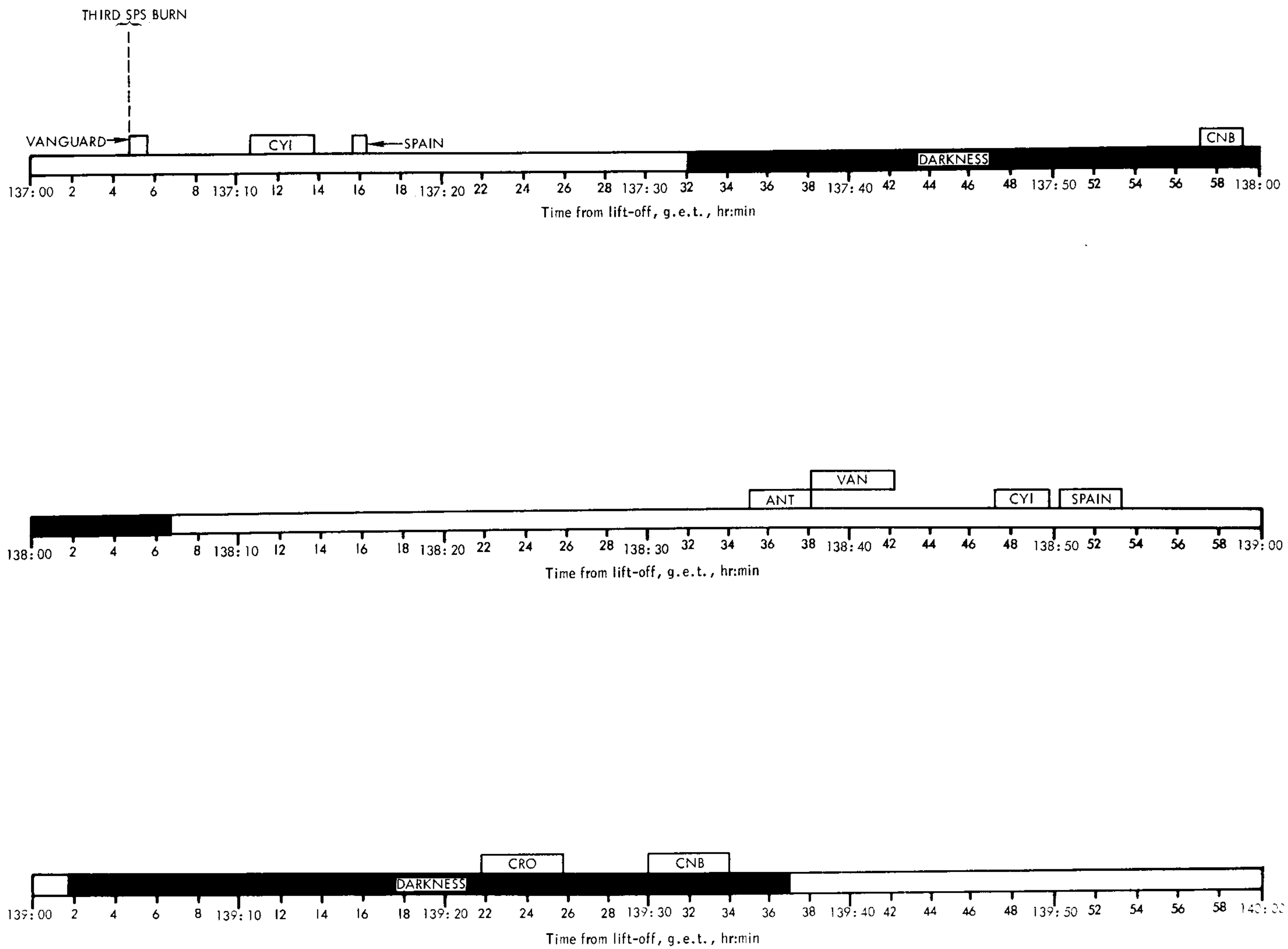
(d) Third period of activity, from 120 hours to 123 hours.

Figure 10. - Continued.



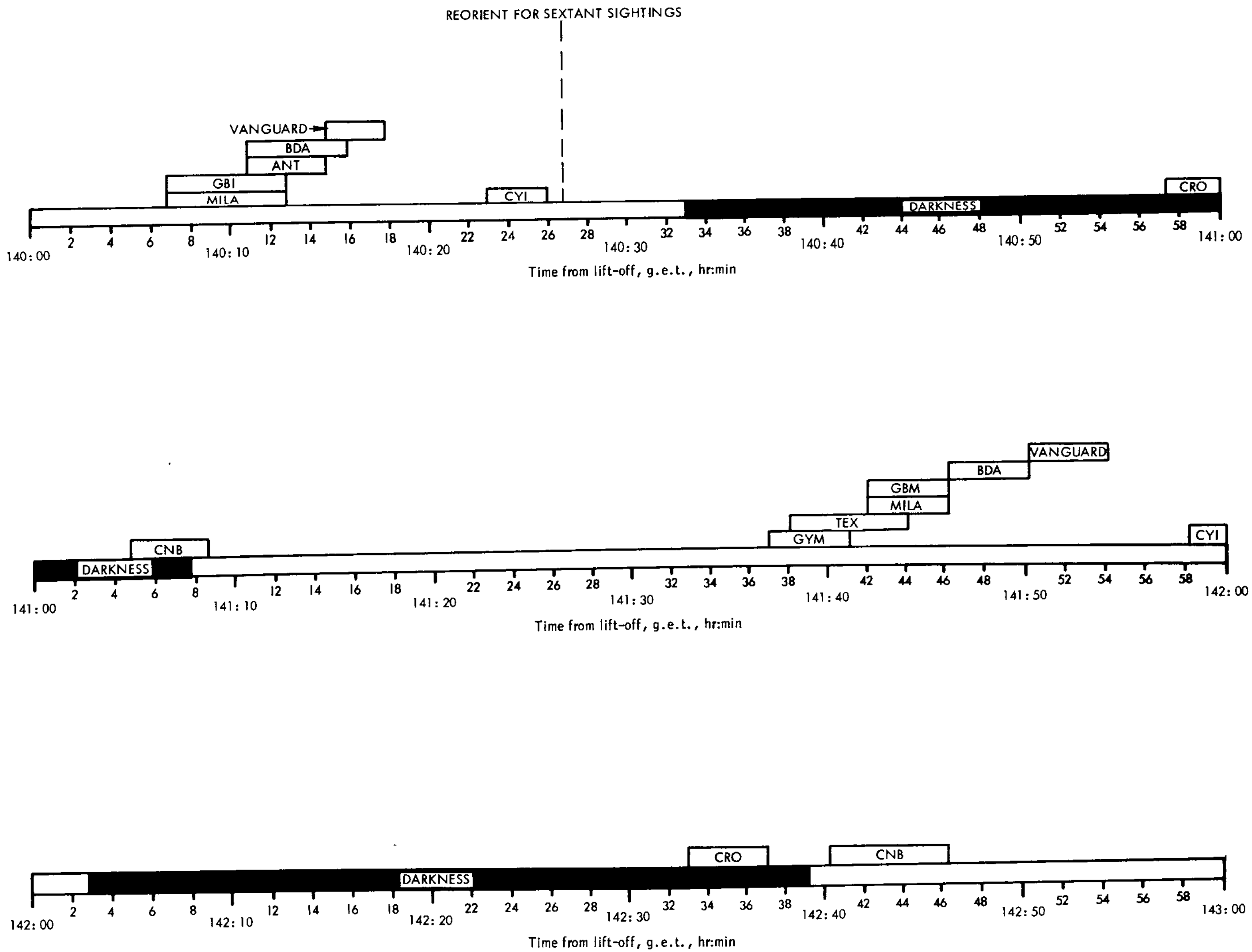
(e) Third period of activity, from 123 hours to 126 hours.

Figure 10. - Continued.



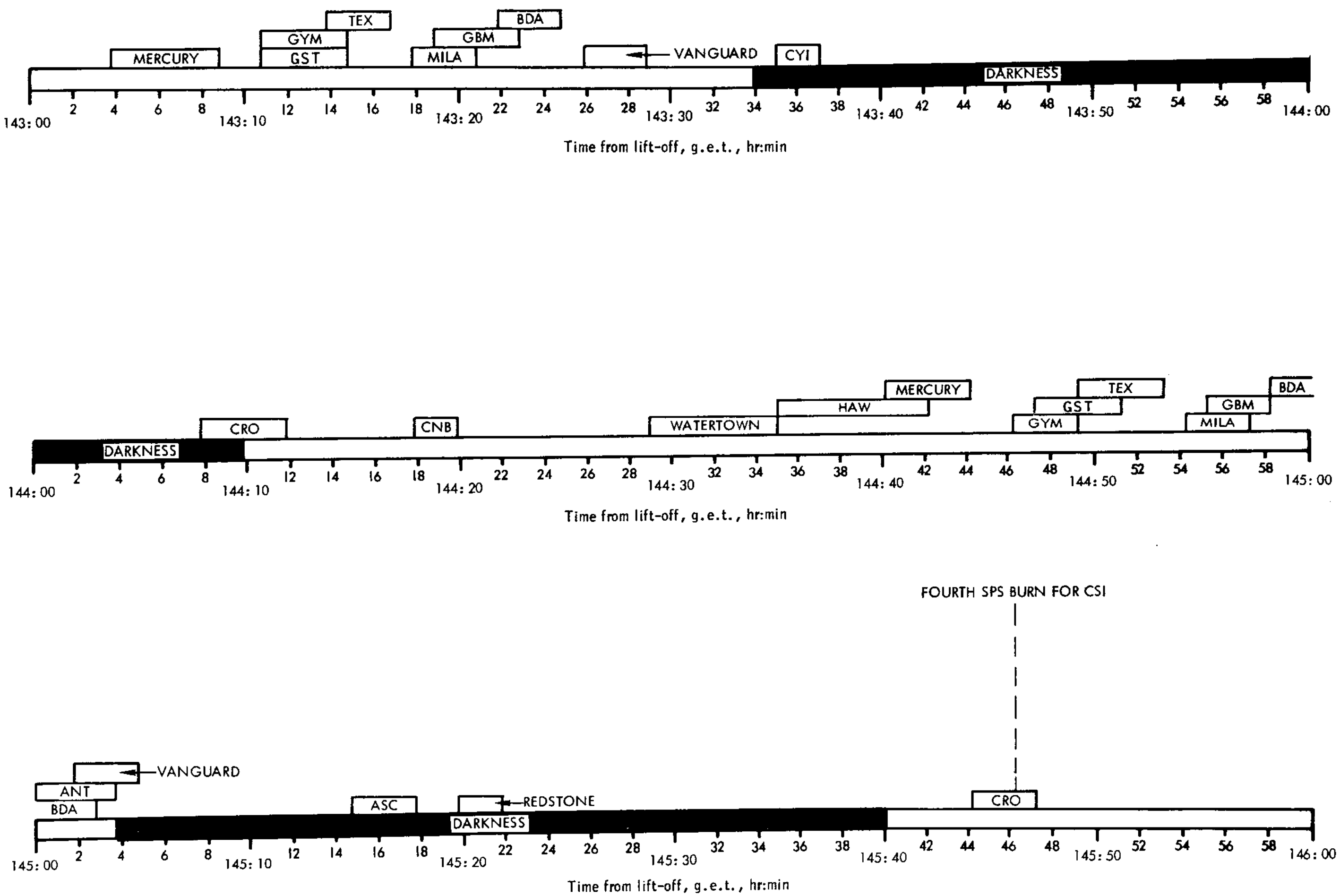
(f) Fourth period of activity, from 137 hours to 140 hours.

Figure 10. - Continued.



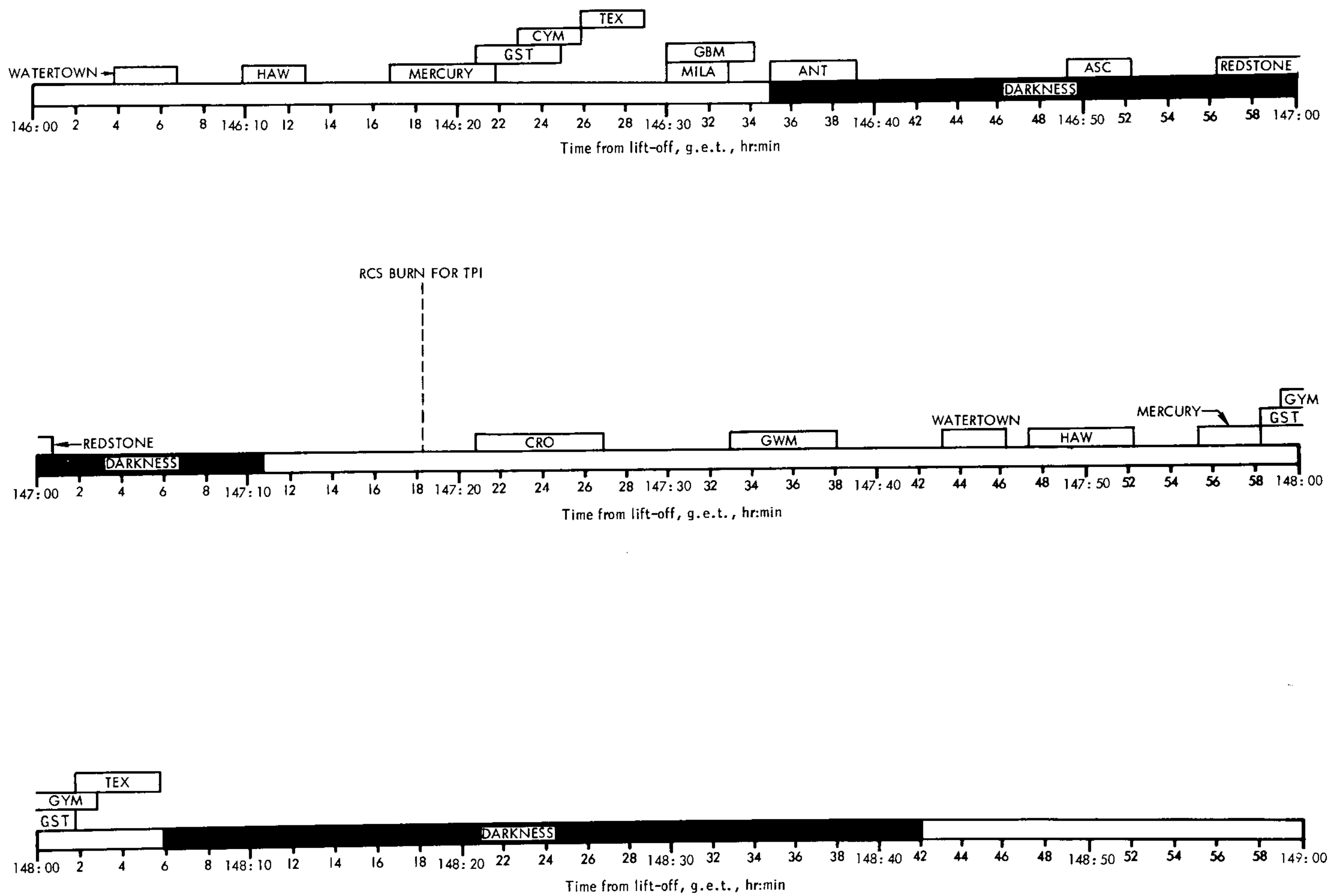
(g) Fourth period of activity, from 140 hours to 143 hours.

Figure 10. - Continued.



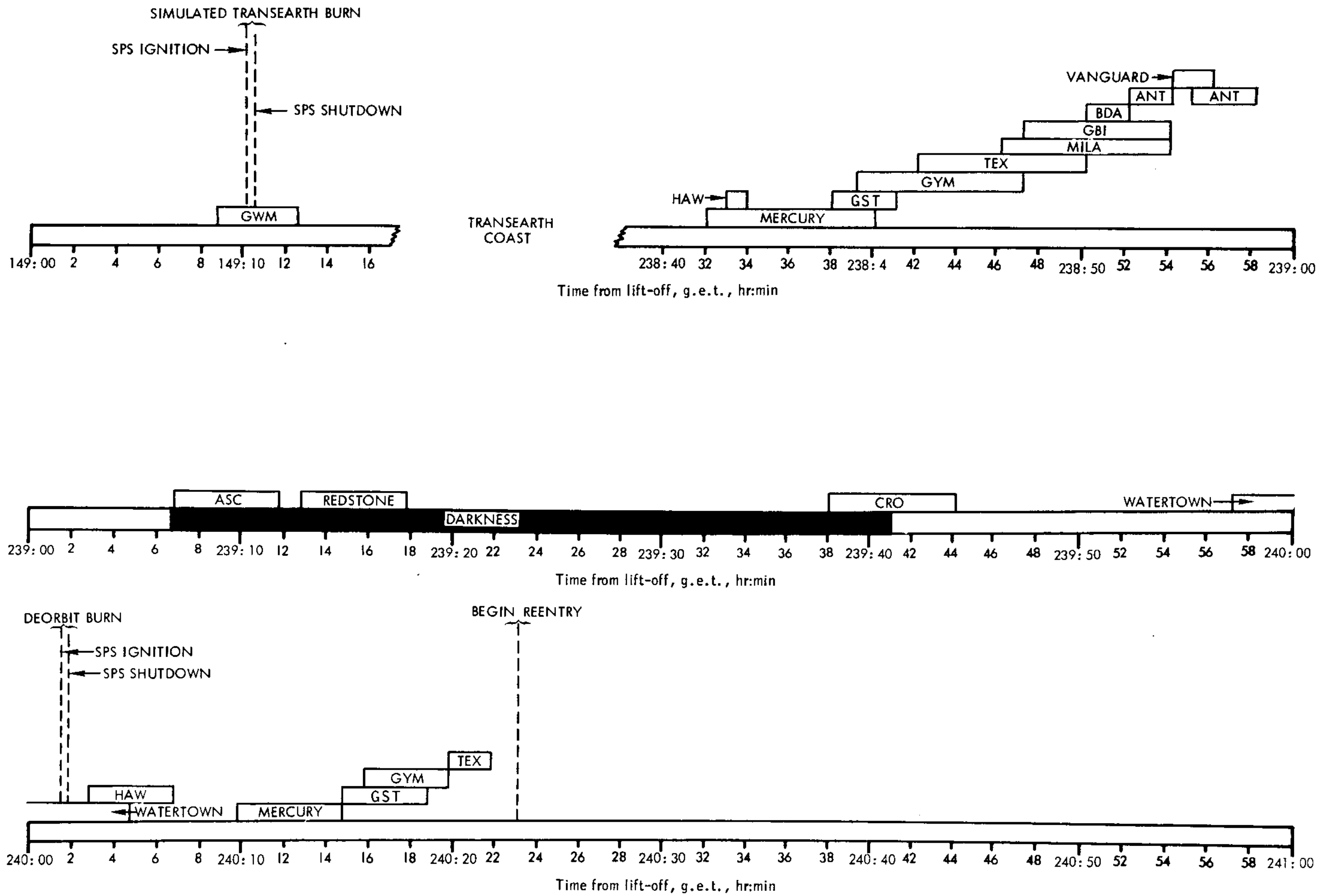
(h) Fourth period of activity, from 143 hours to 146 hours.

Figure 10. - Continued.



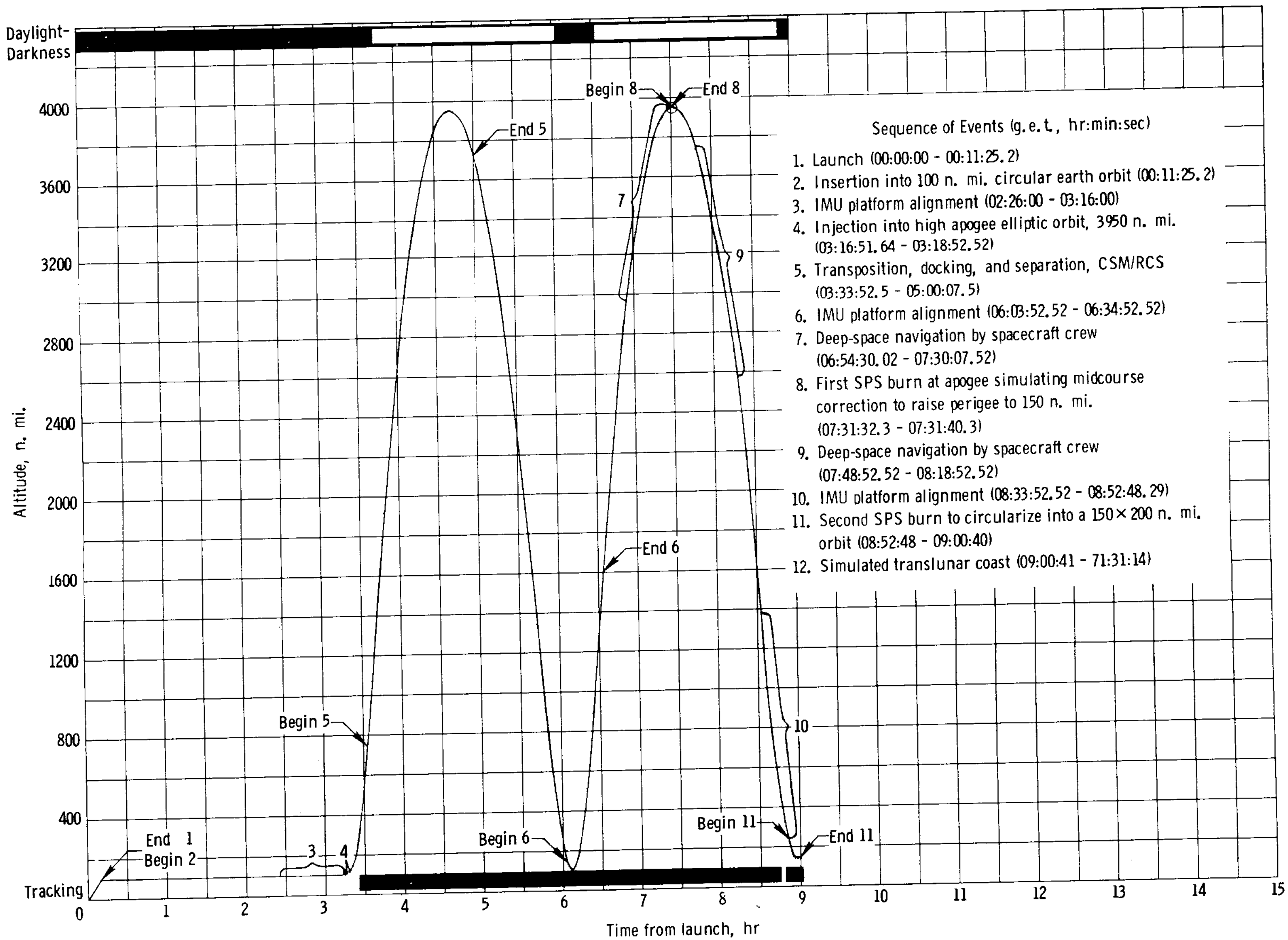
(i) Fourth period of activity, from 146 hours to 149 hours.

Figure 10. - Continued.



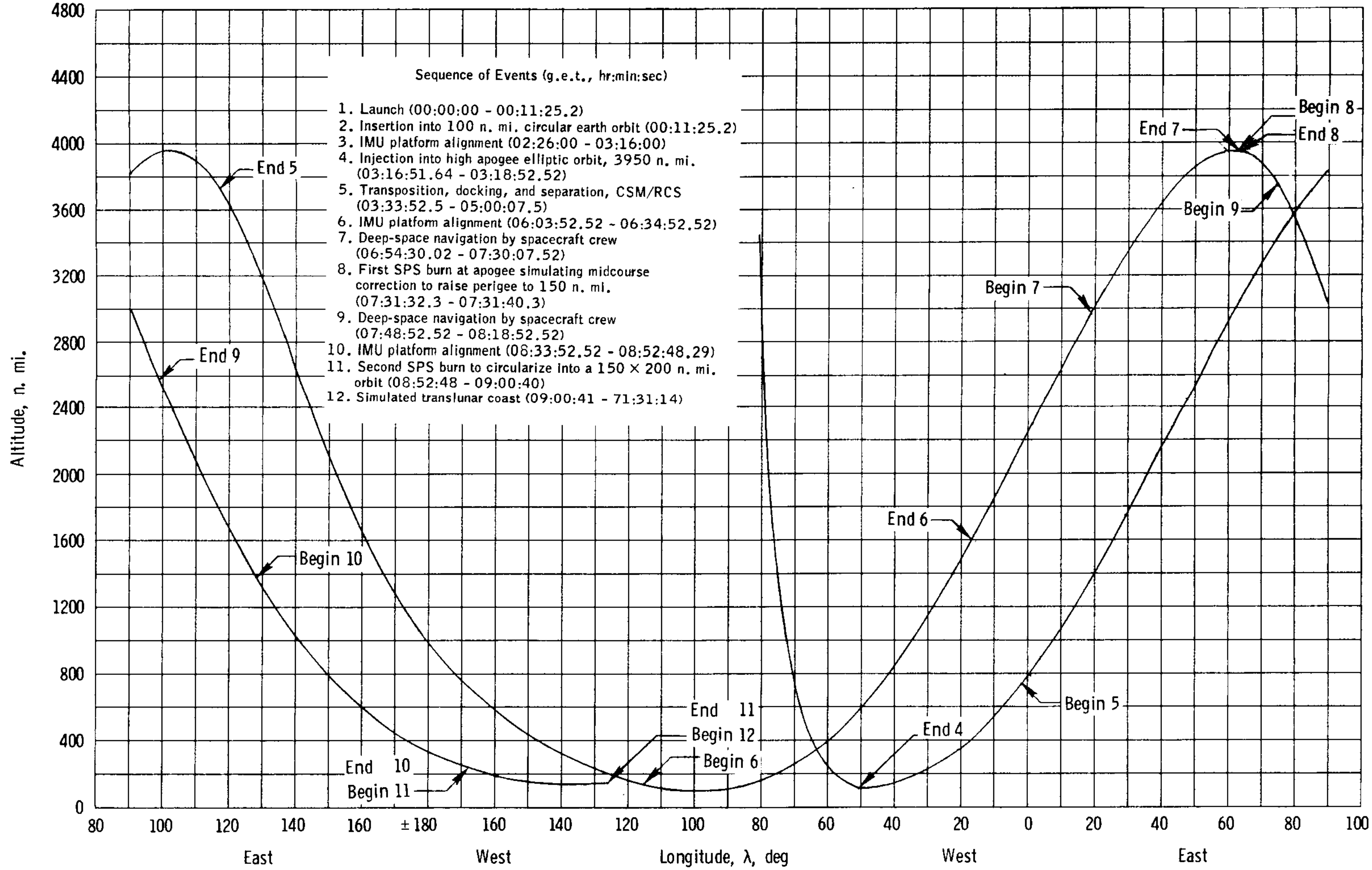
(j) Fourth period of activity, from 149 hours to 241 hours.

Figure 10. - Concluded.



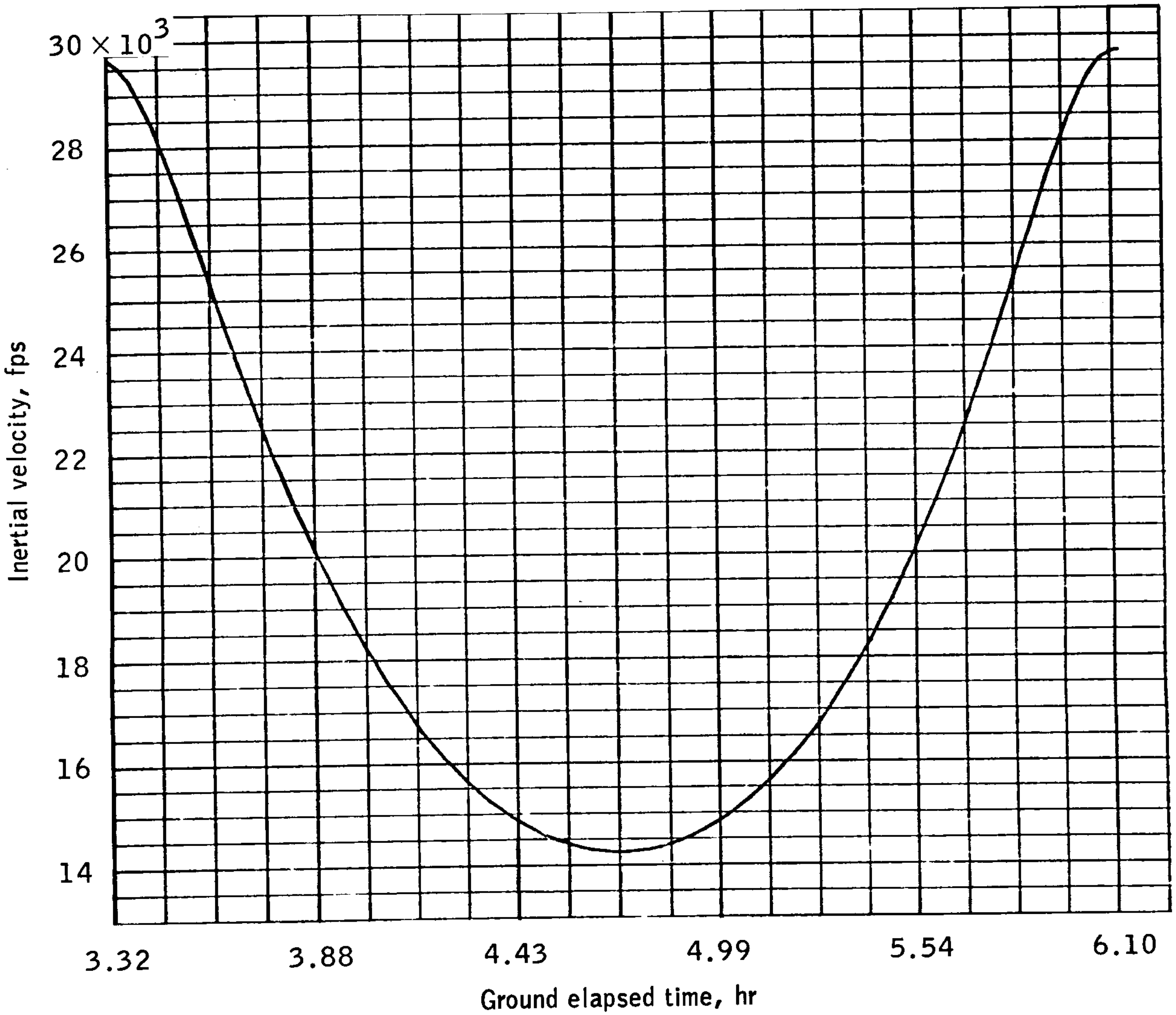
(a) Altitude profile.

Figure 11. - Trajectory parameters during high apogee elliptic orbits.



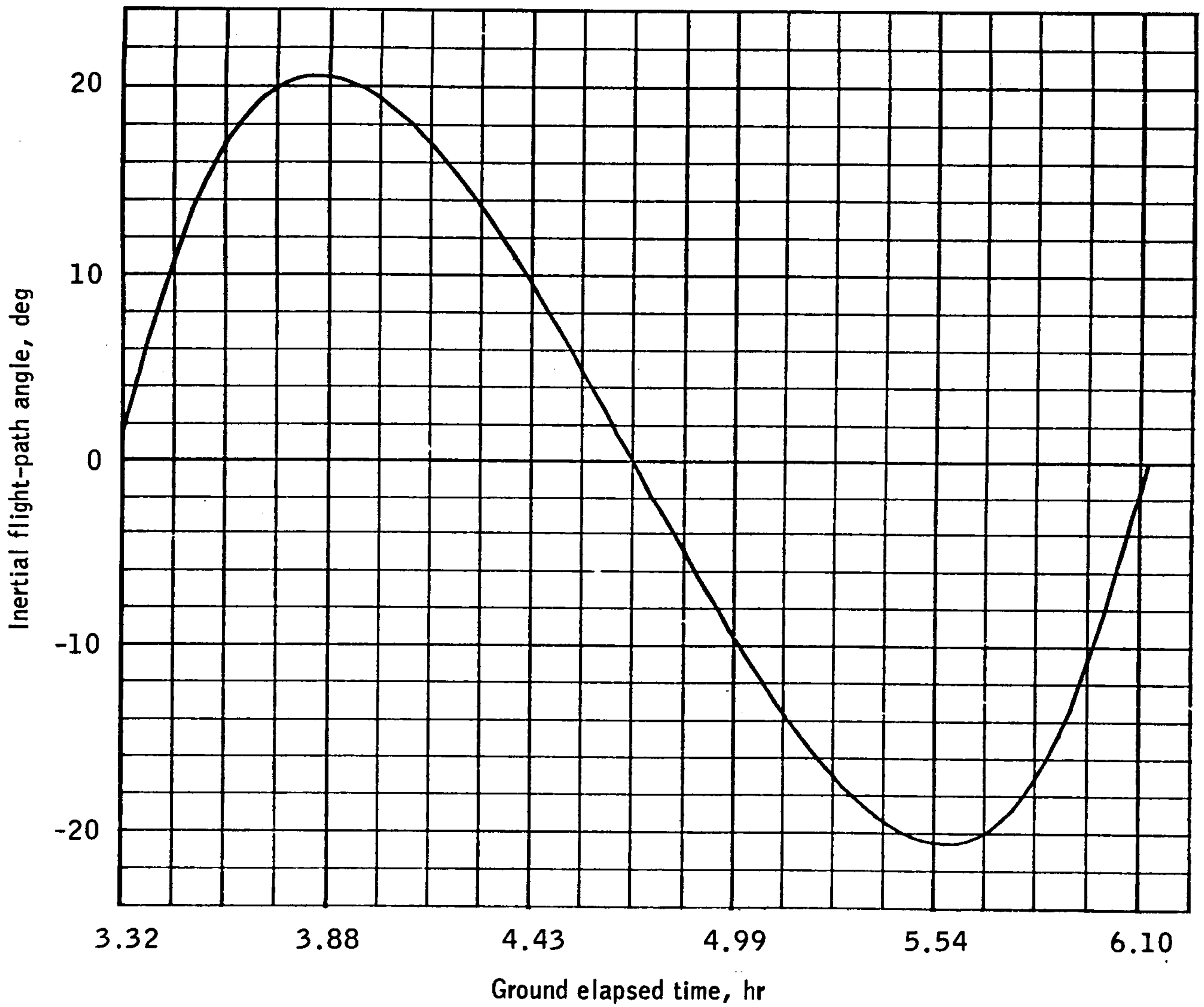
(b) Longitude versus altitude.

Figure 11. - Concluded.



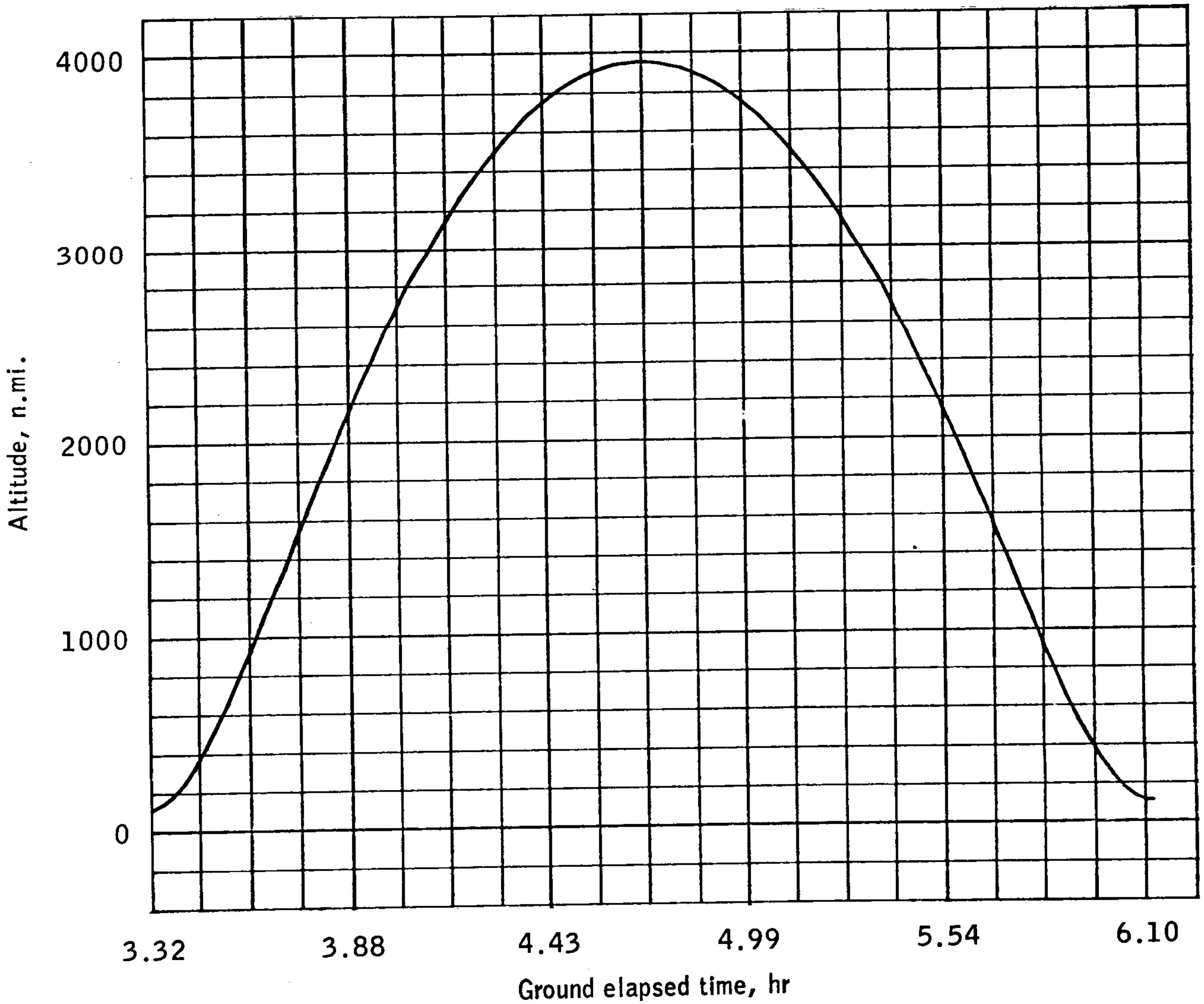
(a) Inertial velocity versus ground elapsed time.

Figure 12.- Time history of trajectory parameters during the first high apogee elliptical orbit.



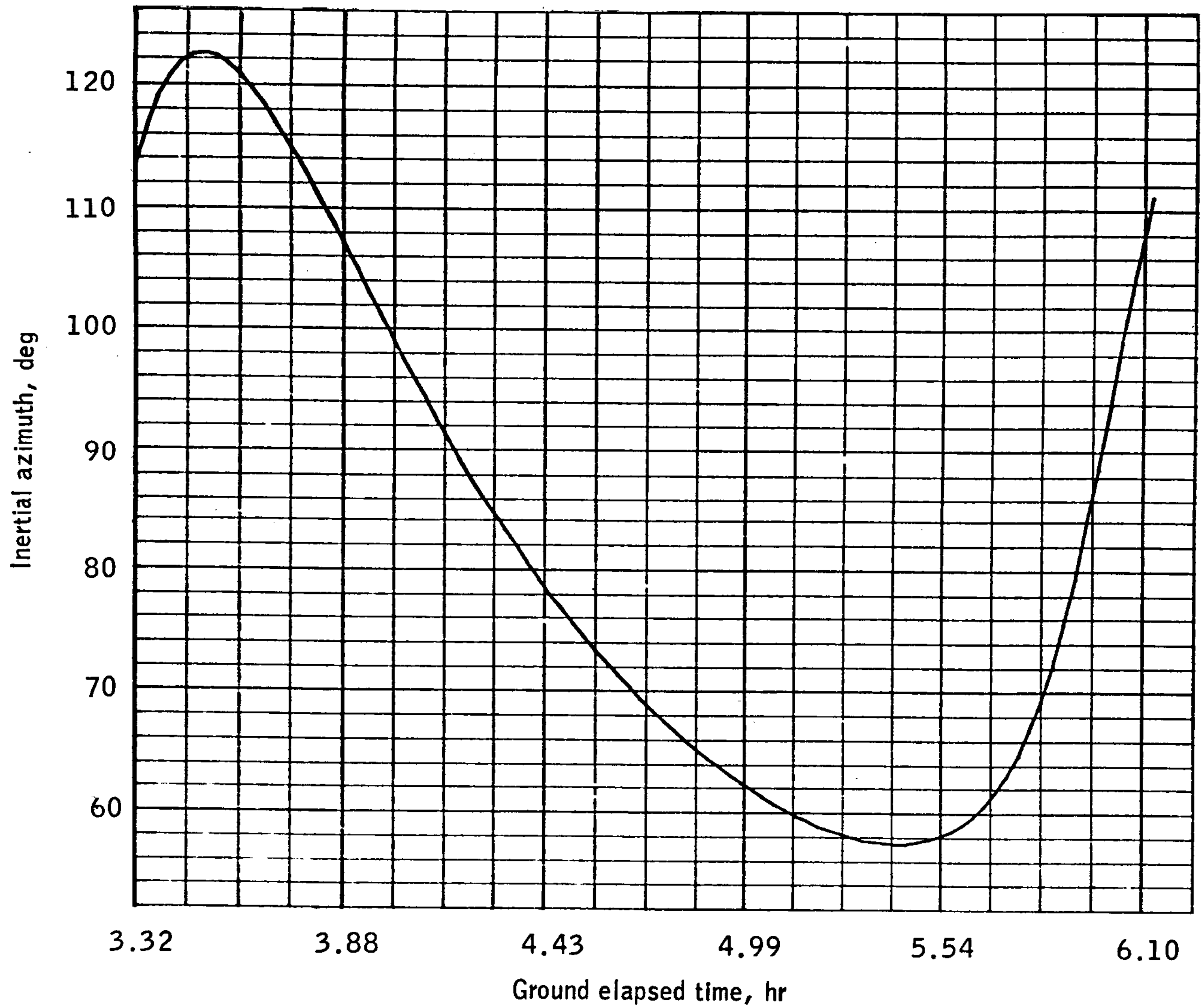
(b) Inertial flight-path angle versus ground elapsed time.

Figure 12.- Continued.



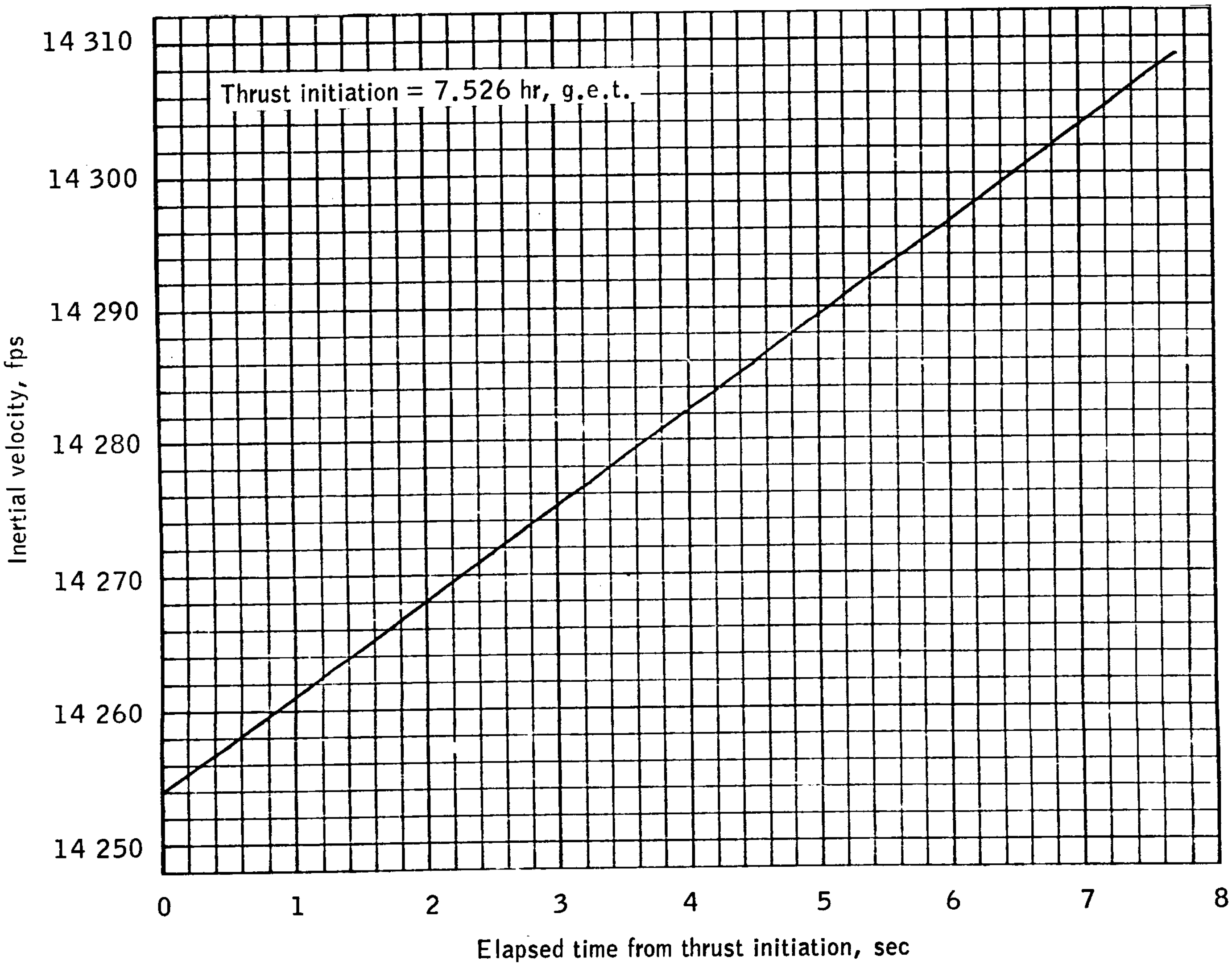
(c) Altitude versus ground elapsed time.

Figure 12.- Continued.



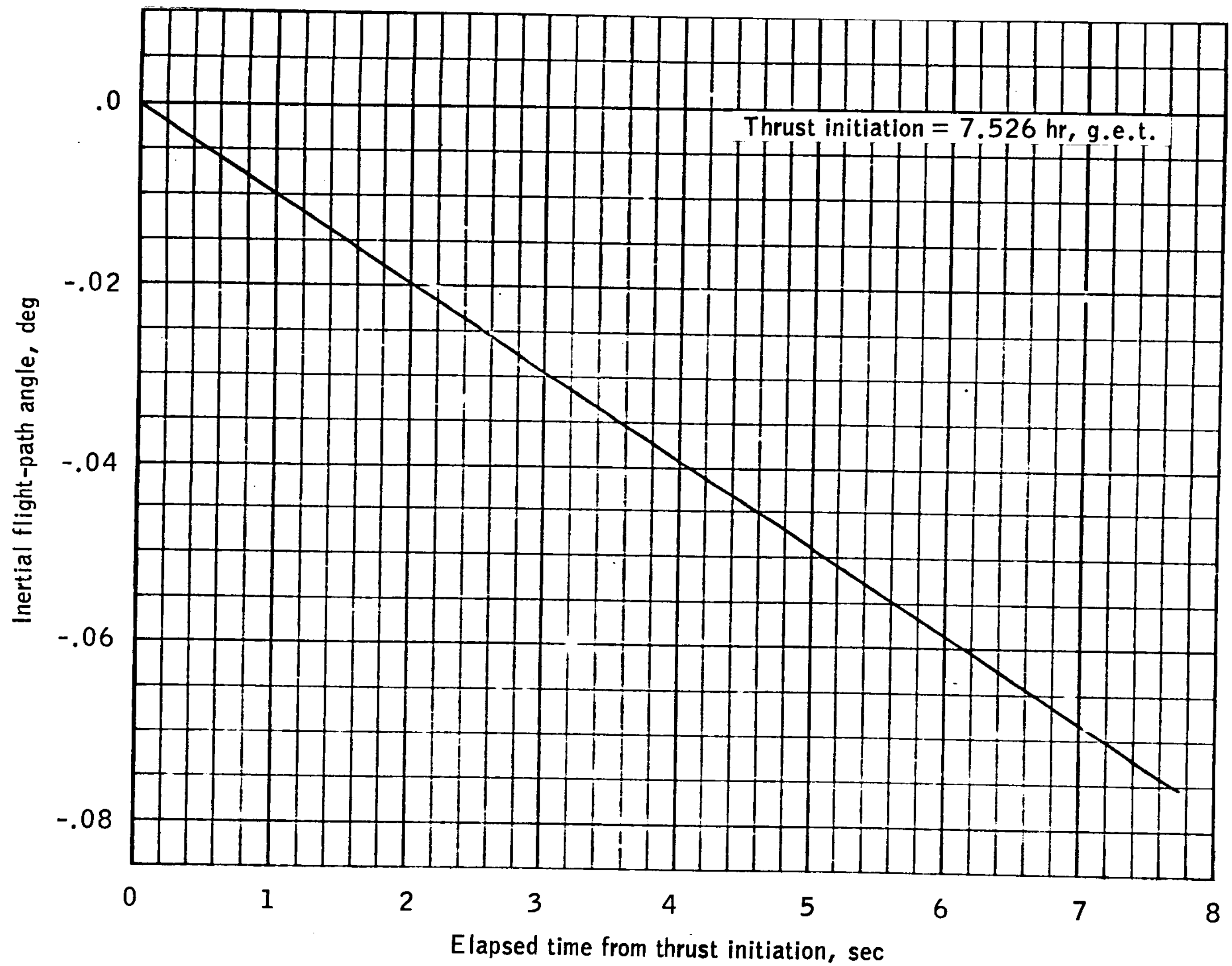
(d) Inertial azimuth versus ground elapsed time.

Figure 12.- Concluded.



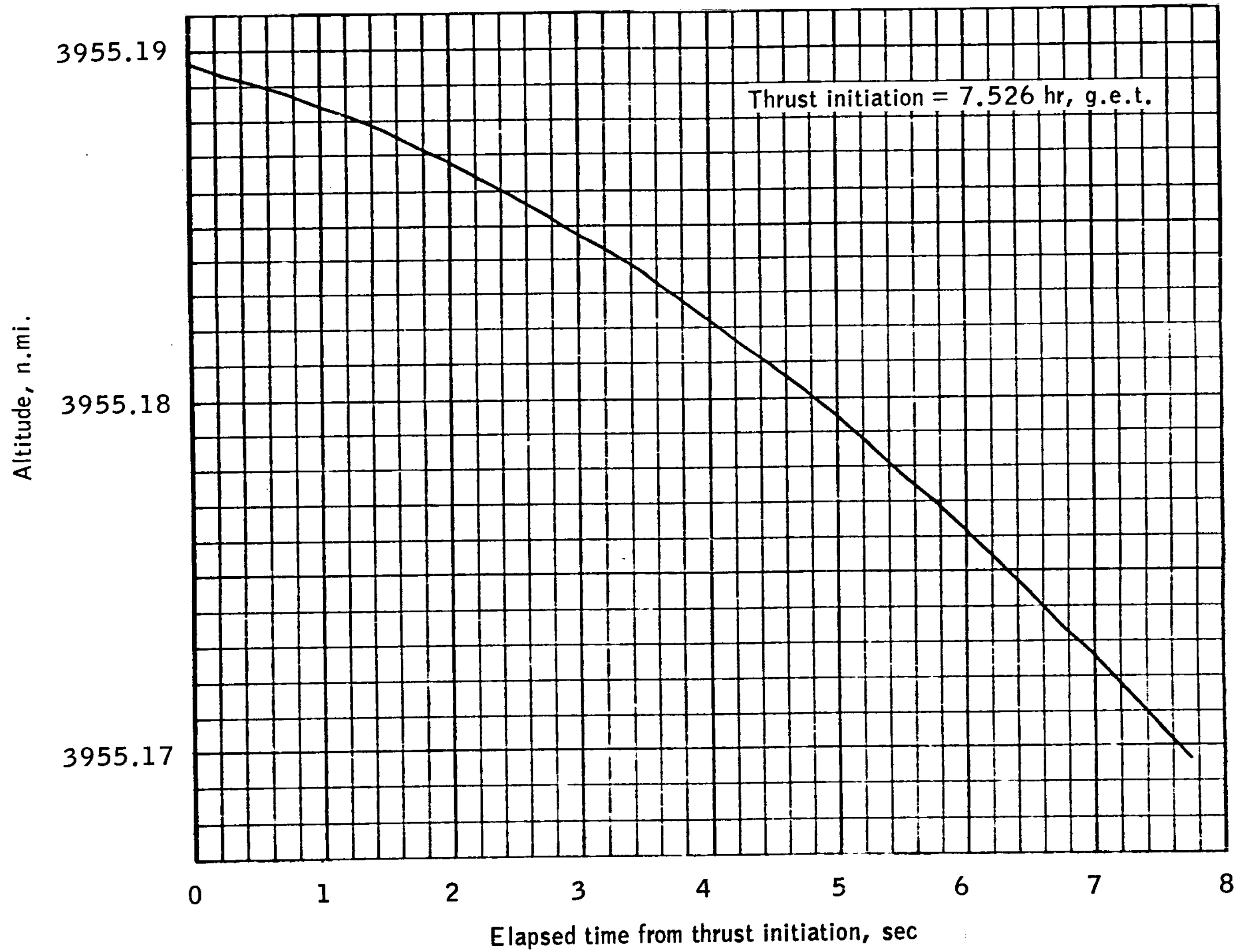
(a) Inertial velocity versus time from thrust initiation.

Figure 13.- Time history of trajectory parameters during the SPS burn to raise perigee to 150 n.mi. (Event 8)



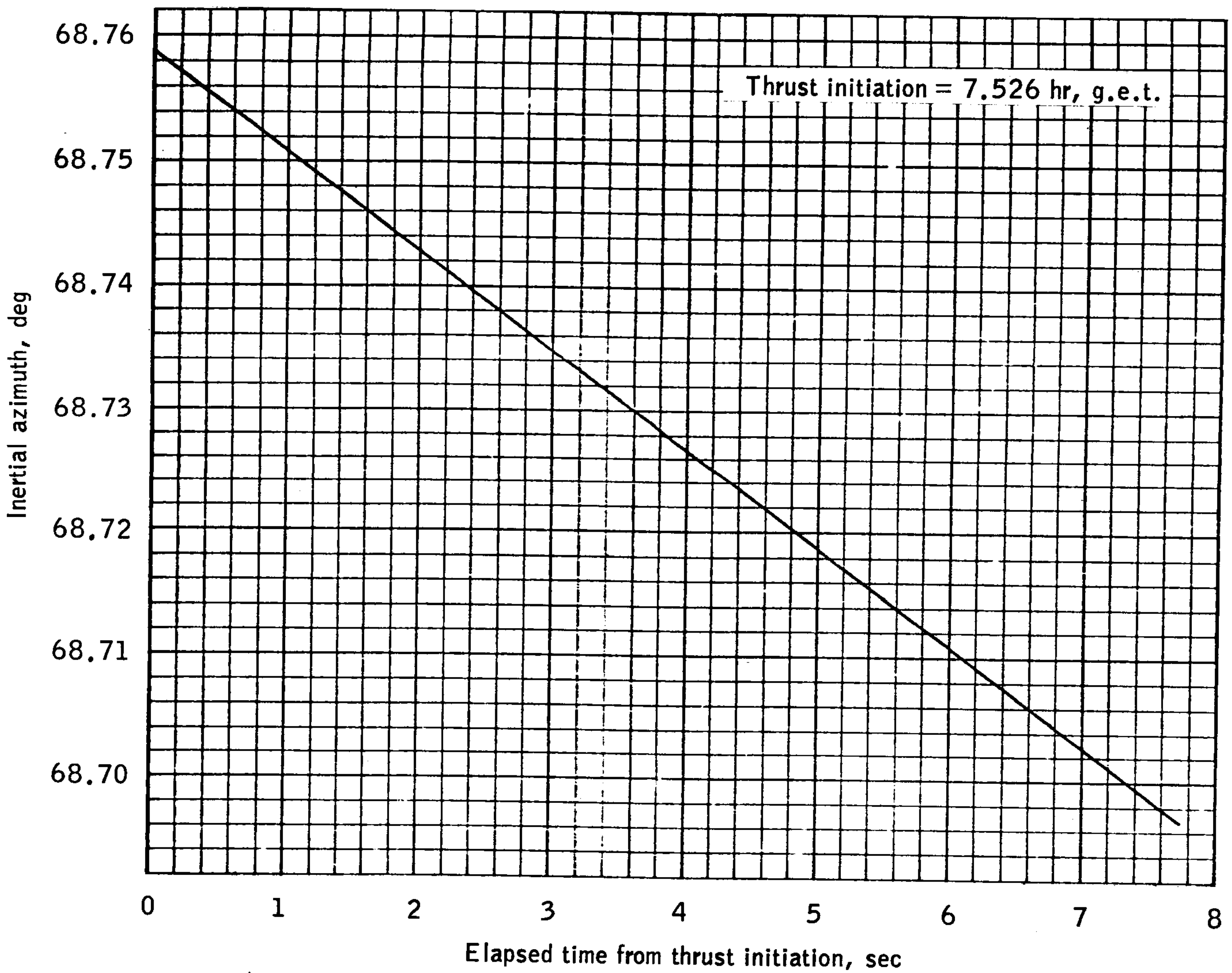
(b) Inertial flight-path angle versus time from thrust initiation.

Figure 13.- Continued.



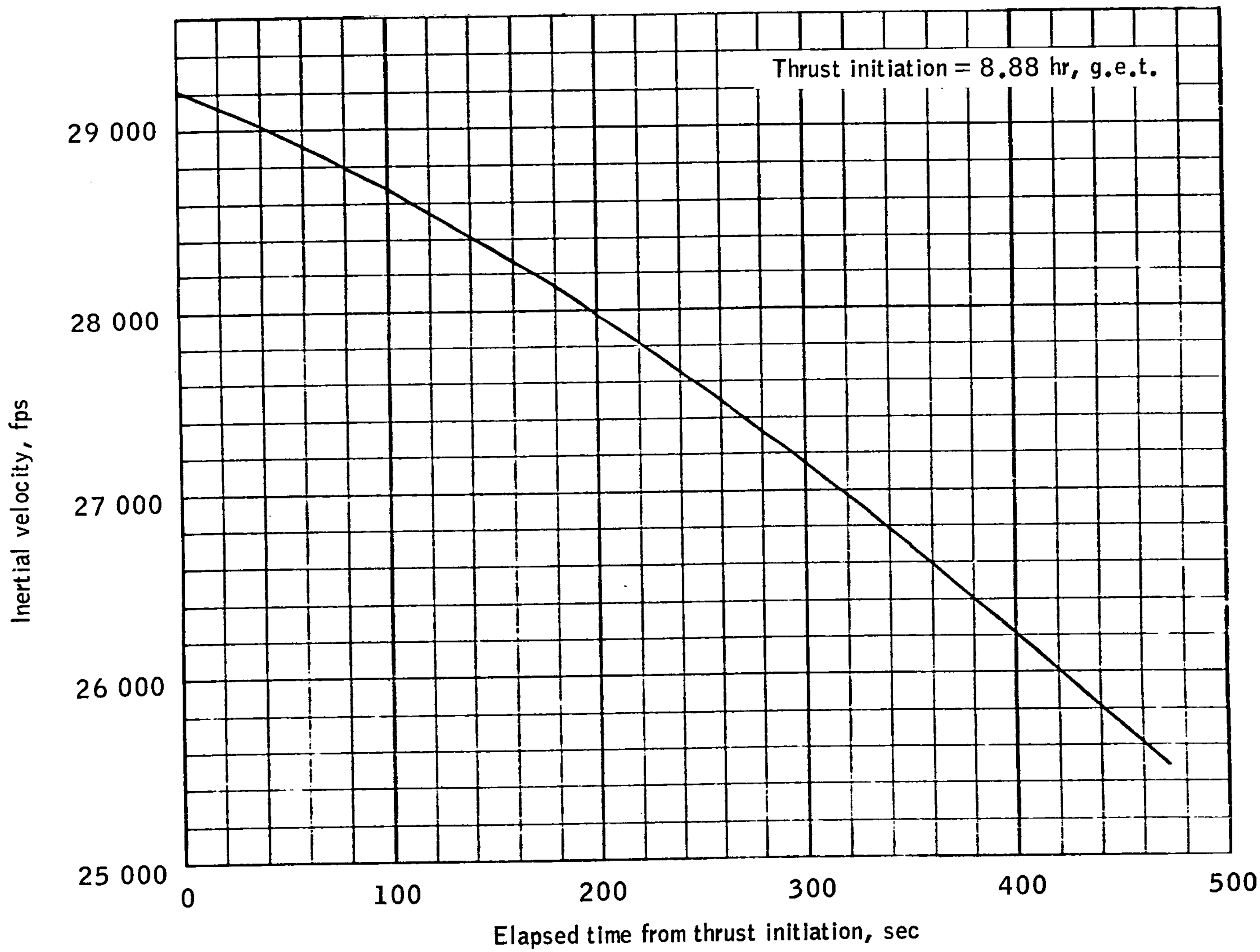
(c) Altitude versus time from thrust initiation.

Figure 13.- Continued.



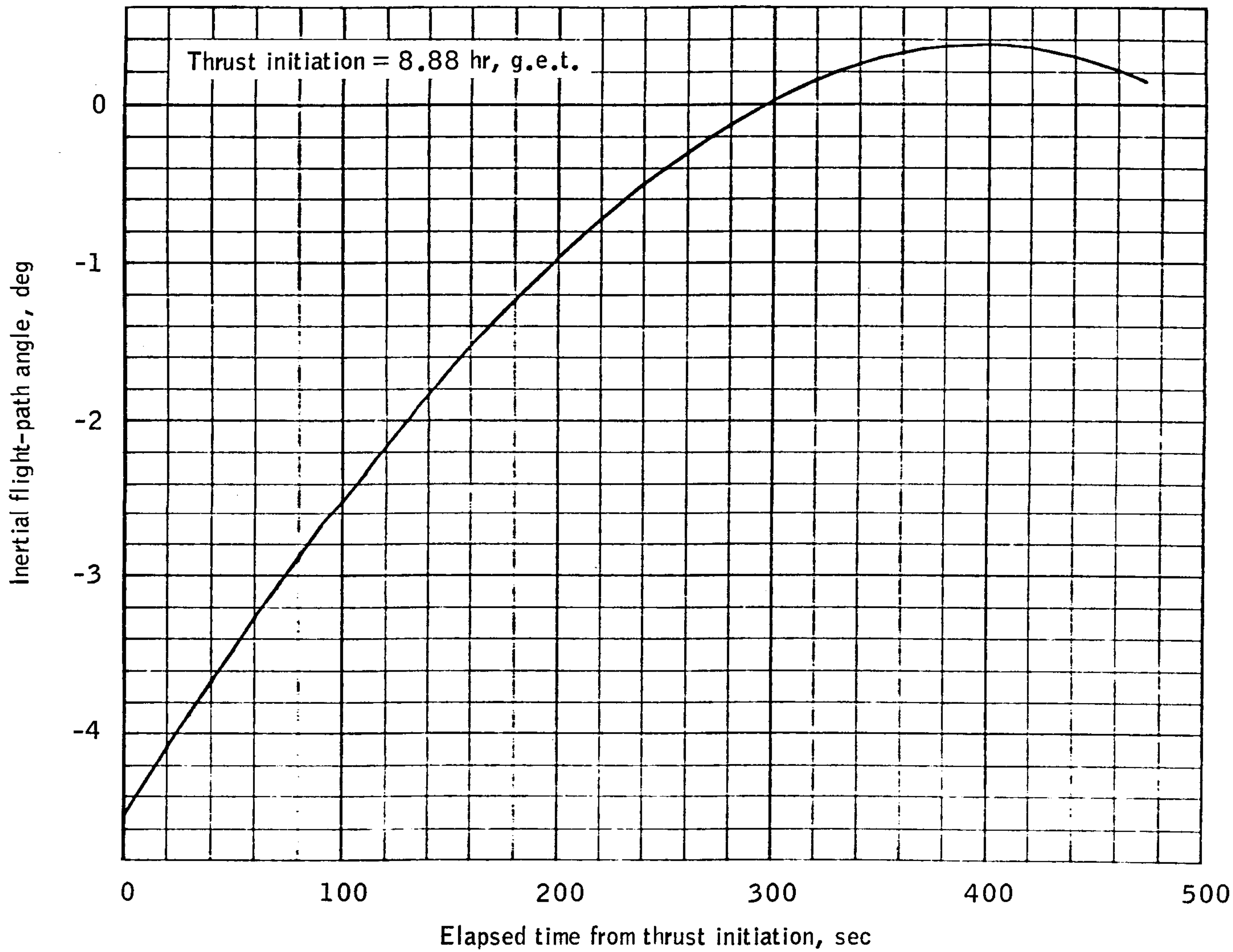
(d) Inertial azimuth versus time from thrust initiation.

Figure 13.- Concluded.



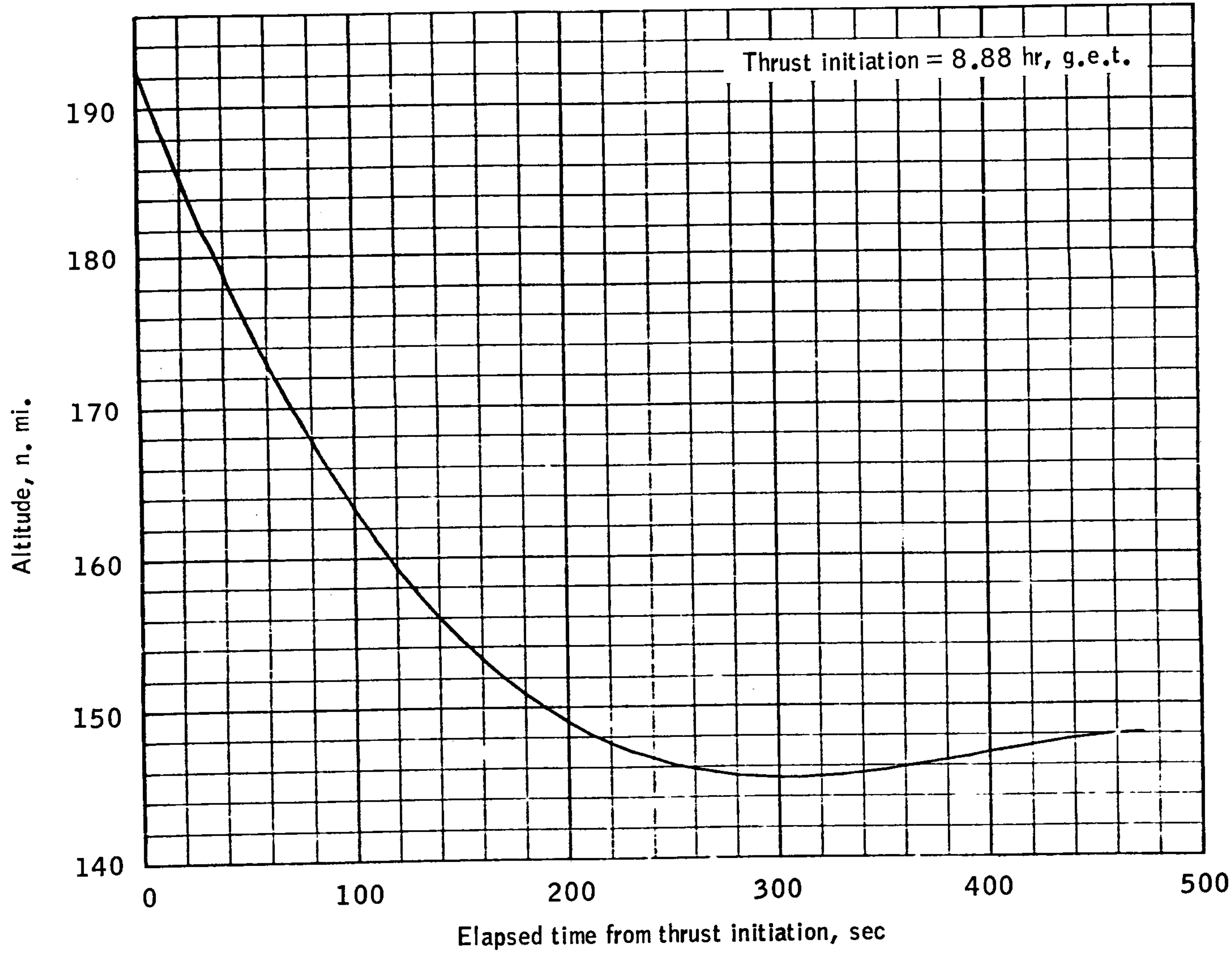
(a) Inertial velocity versus time from thrust initiation.

Figure 14.- Time history of trajectory parameters during the SPS burn to lower apogee to 200 n. mi. (Event 11)



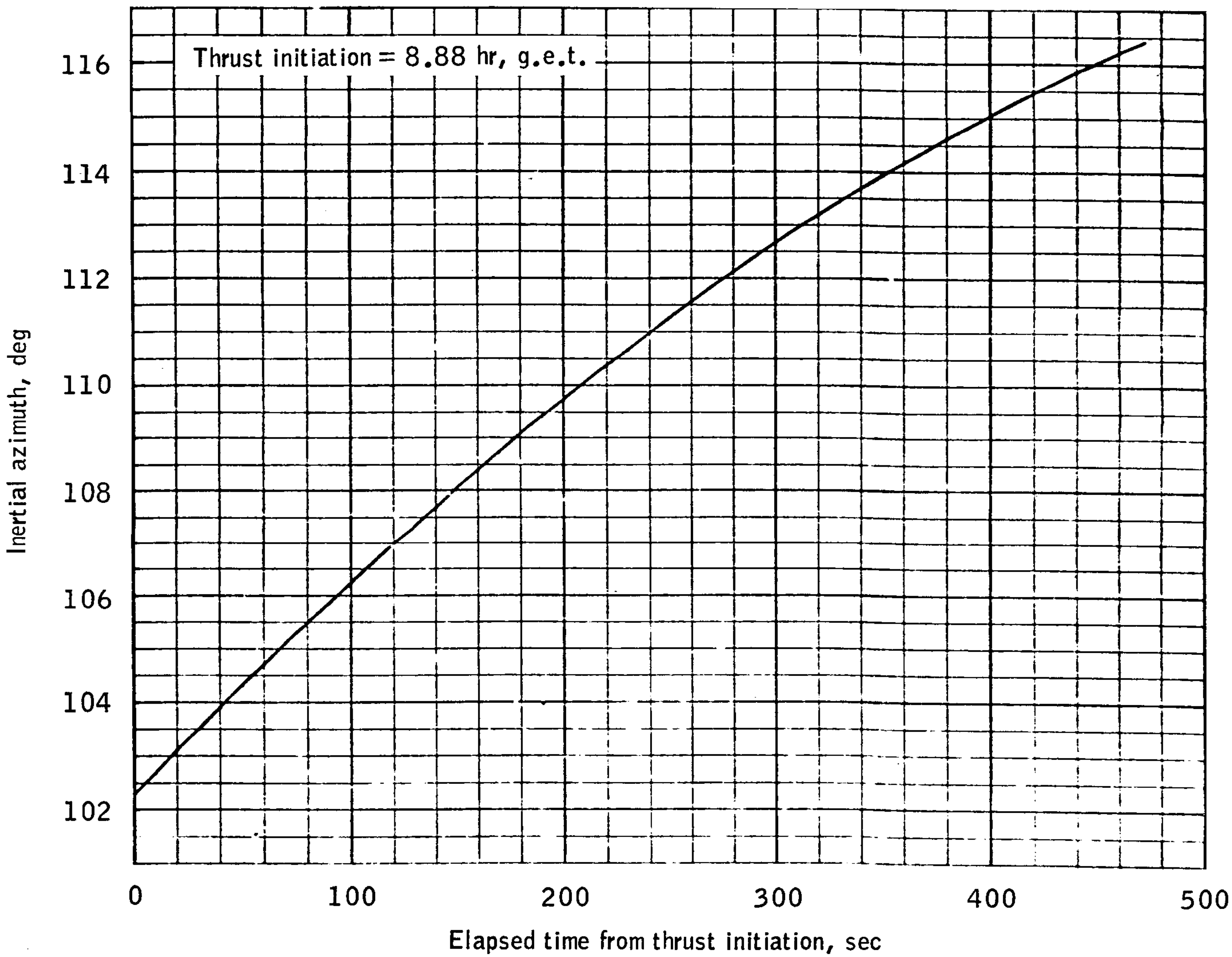
(b) Inertial flight-path angle versus time from thrust initiation.

Figure 14.- Continued.



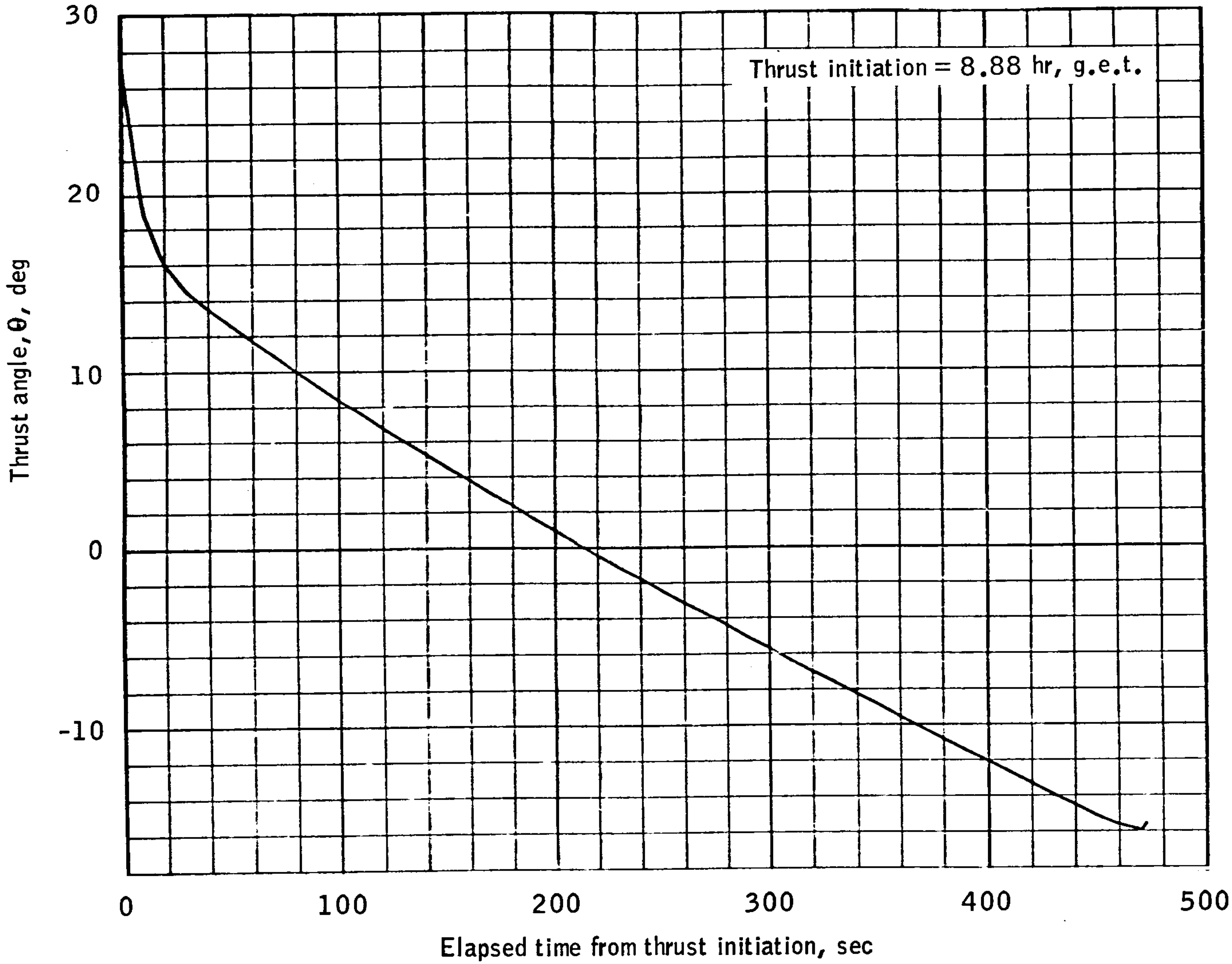
(c) Altitude versus time from thrust initiation.

Figure 14.- Continued.



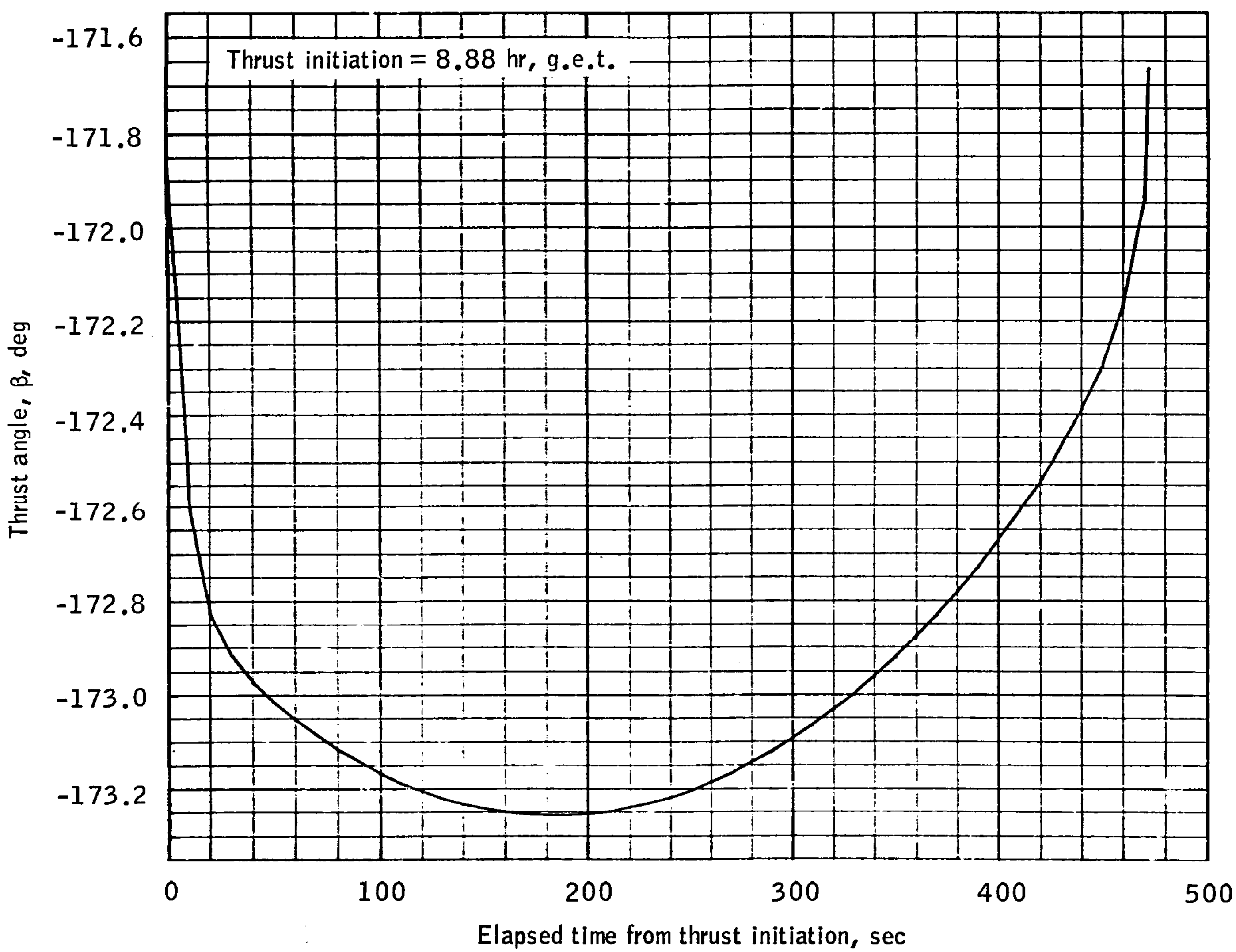
(d) Inertial azimuth versus time from thrust initiation.

Figure 14.- Continued.



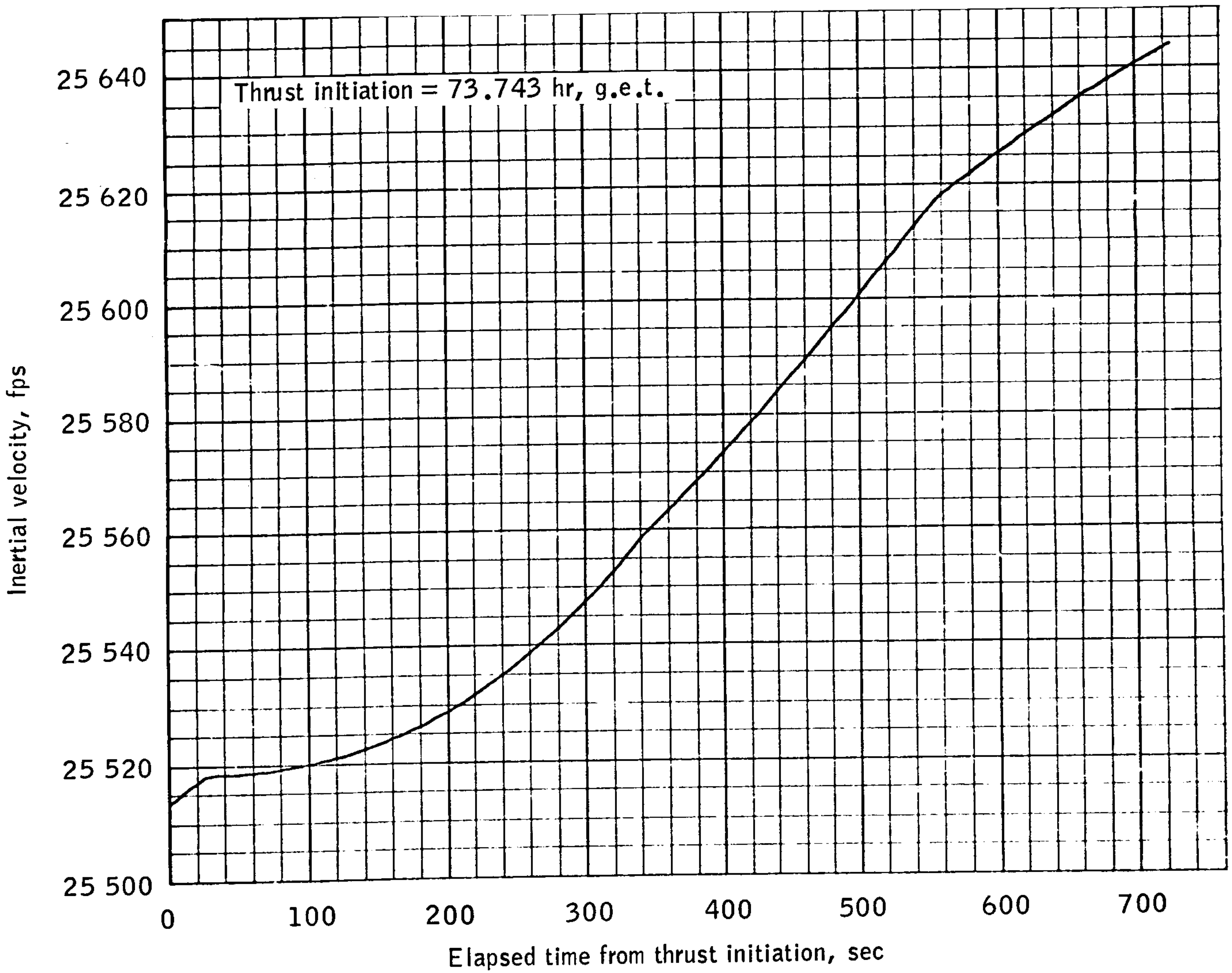
(e) Thrust angle theta versus time from thrust initiation.

Figure 14.- Continued.



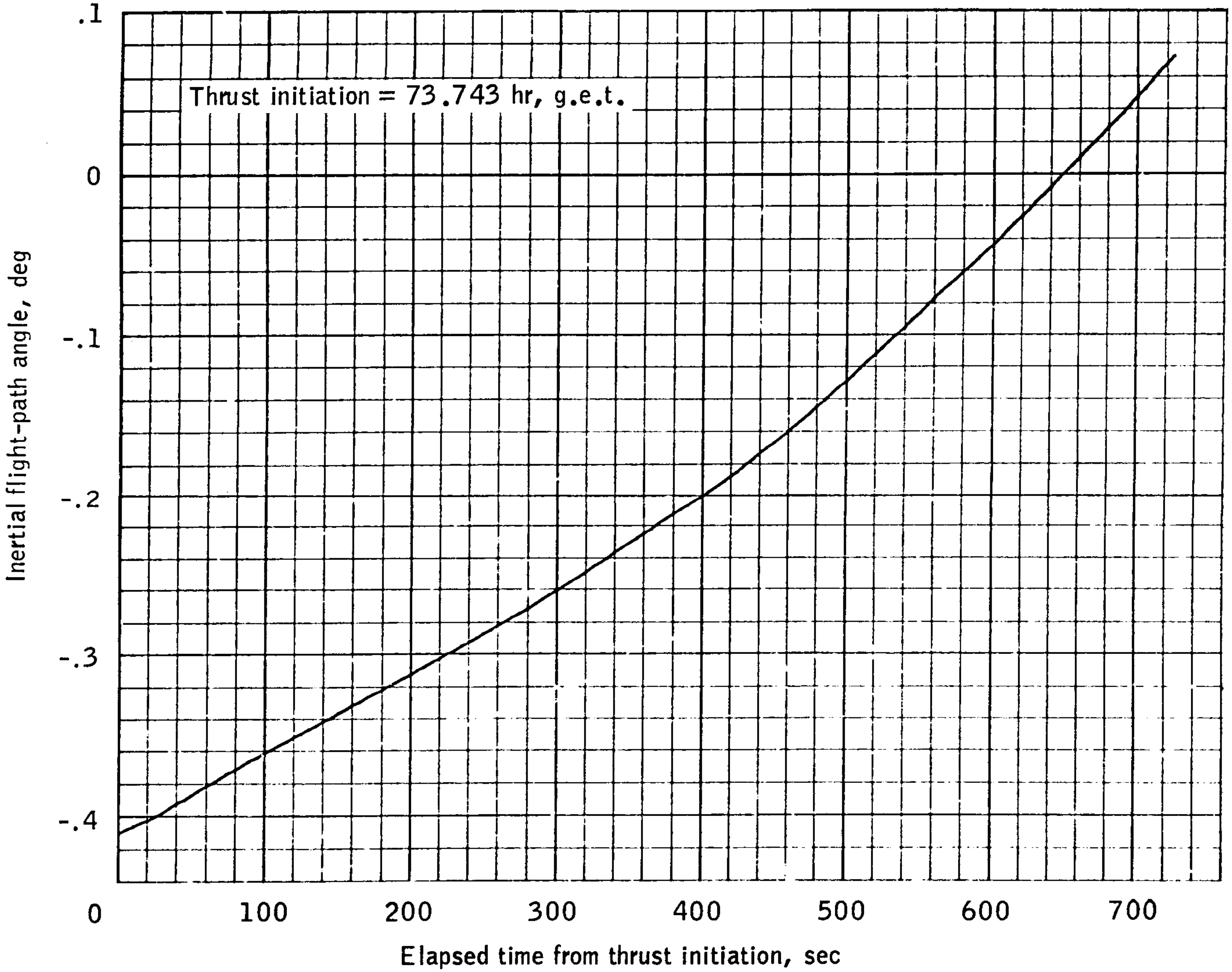
(f) Thrust angle beta versus time from thrust initiation.

Figure 14.- Concluded.



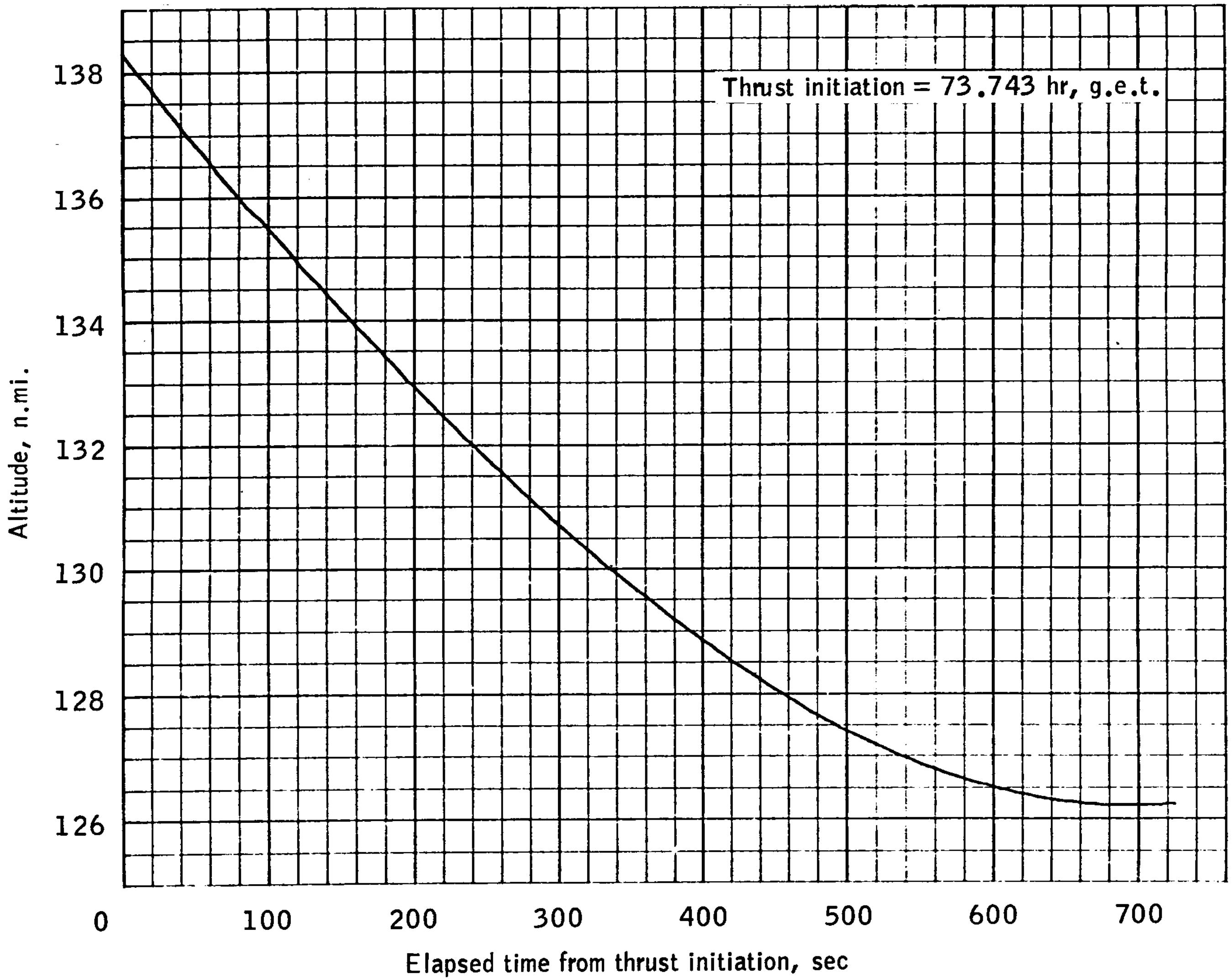
(a) Inertial velocity versus time from thrust initiation.

Figure 15.- Time history of trajectory parameters during the second docked DPS burn. (Event 14)



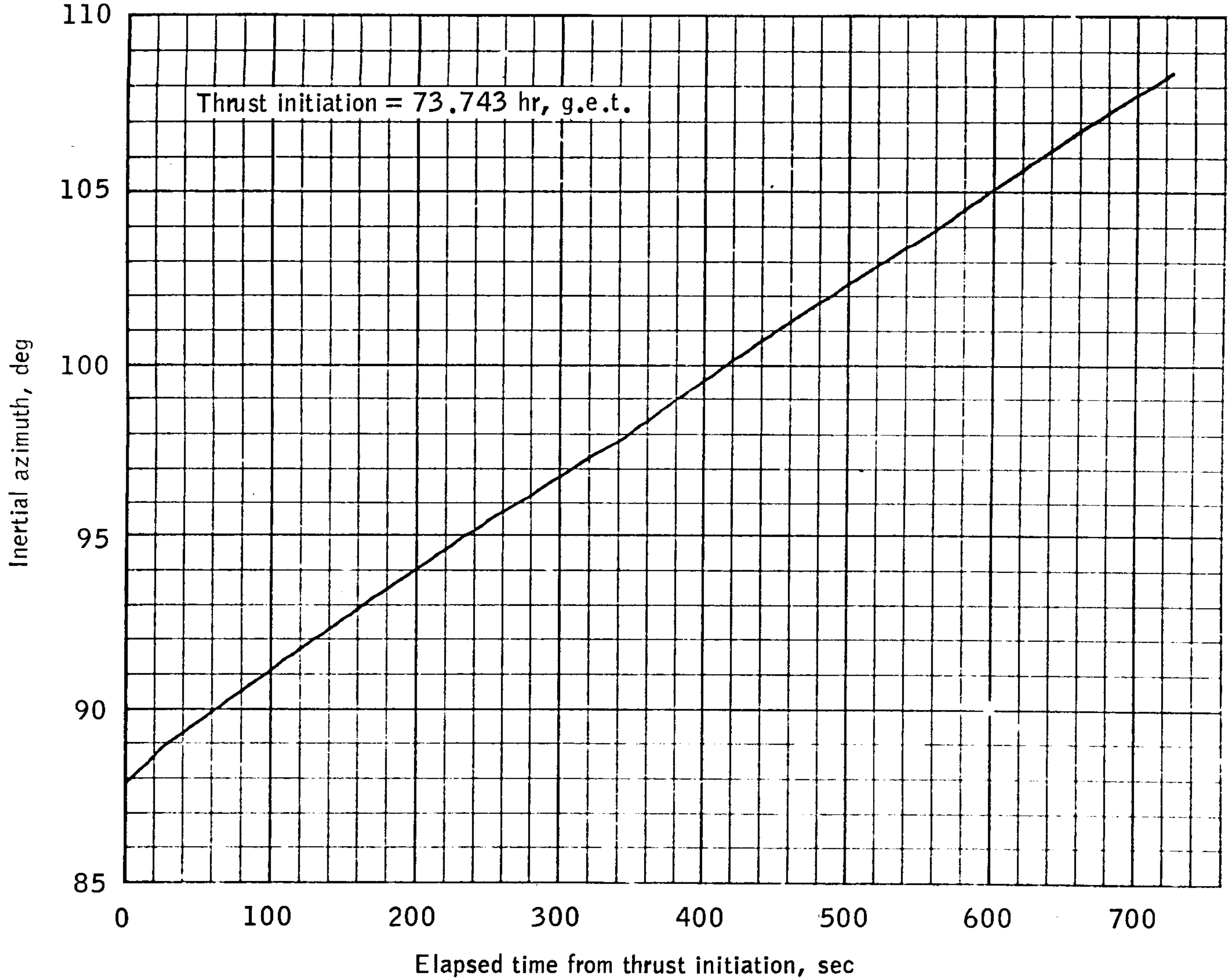
(b) Inertial flight-path angle versus time from thrust initiation.

Figure 15.- Continued.



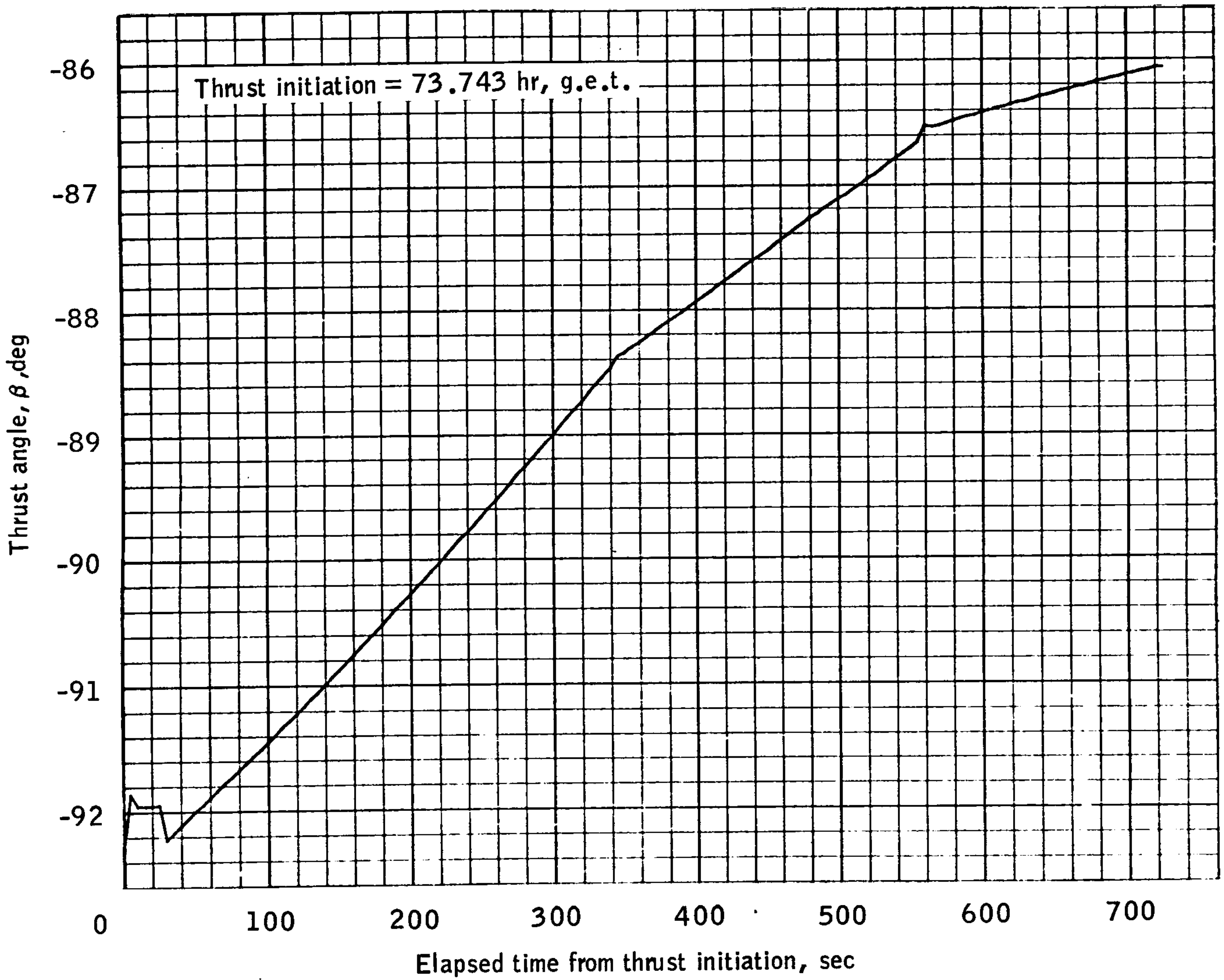
(c) Altitude versus time from thrust initiation.

Figure 15.- Continued.



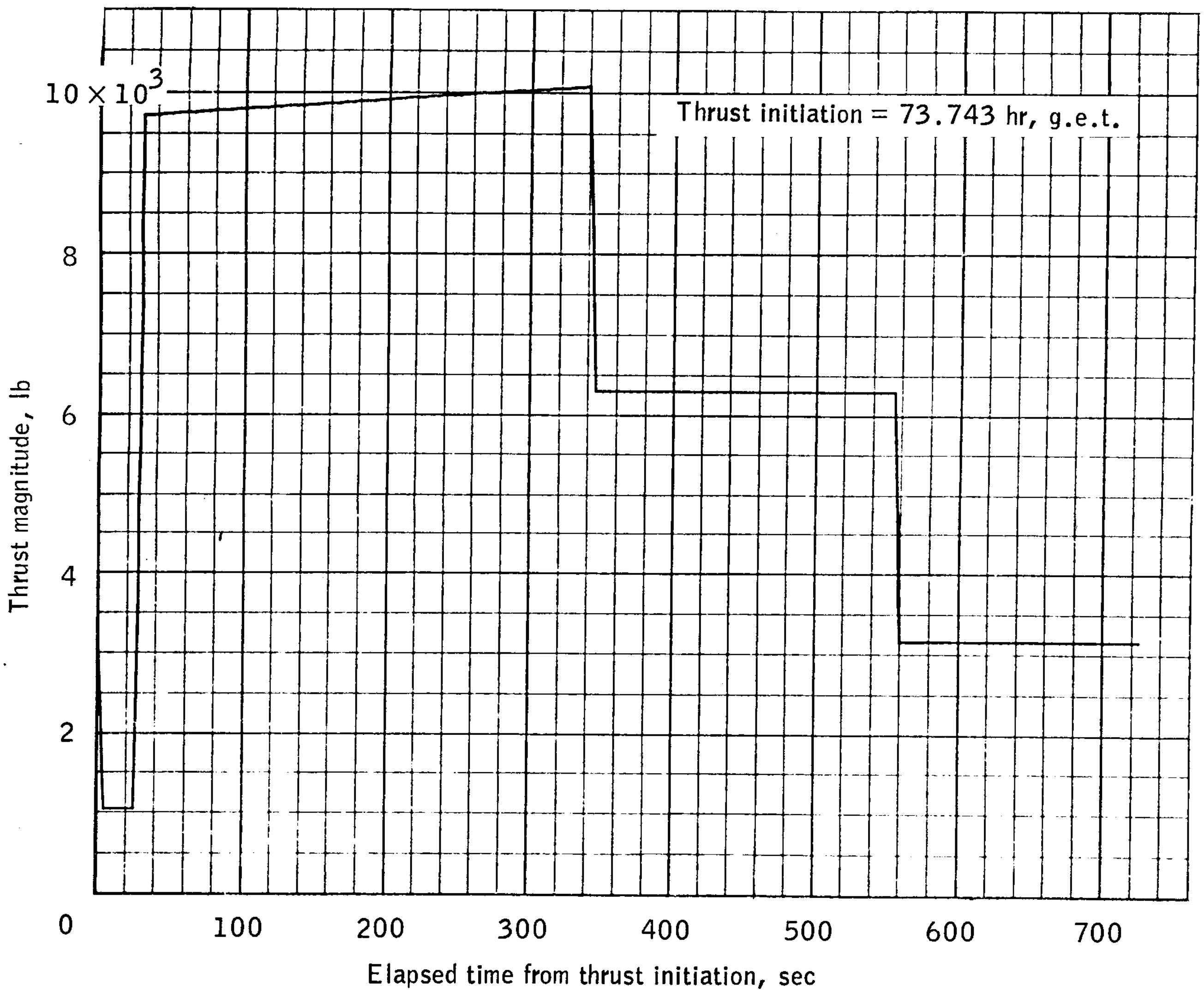
(d) Inertial azimuth versus time from thrust initiation.

Figure 15.- Continued.



(e) Thrust angle beta versus time from thrust initiation.

Figure 15.- Continued.



(f) Thrust magnitude versus time from thrust initiation.

Figure 15.- Concluded.

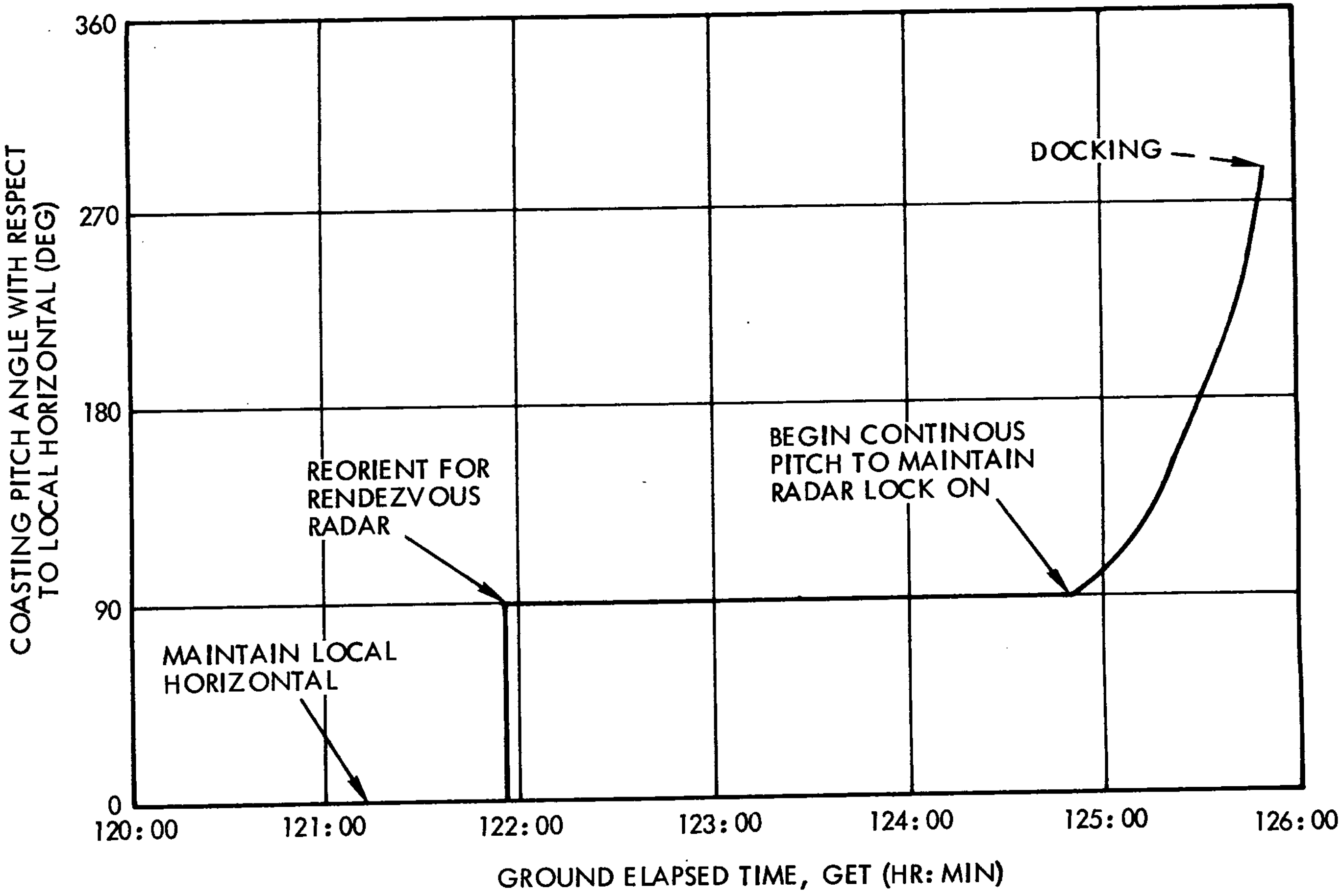


Figure 16.- Body pitch attitude of LM during LM active rendezvous.

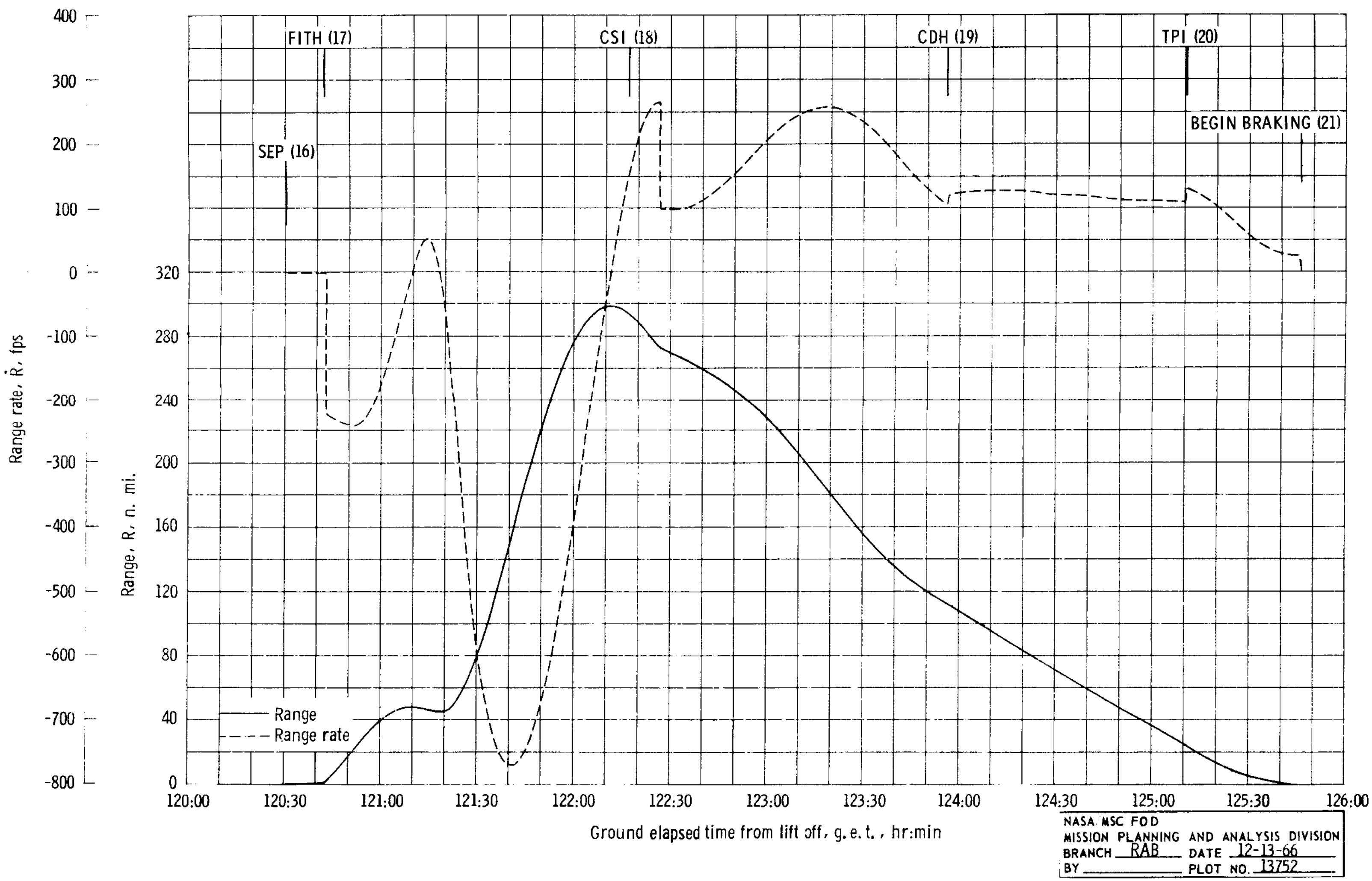


Figure 17. - Time history of range and range rate between LM and CSM during LM-active rendezvous..

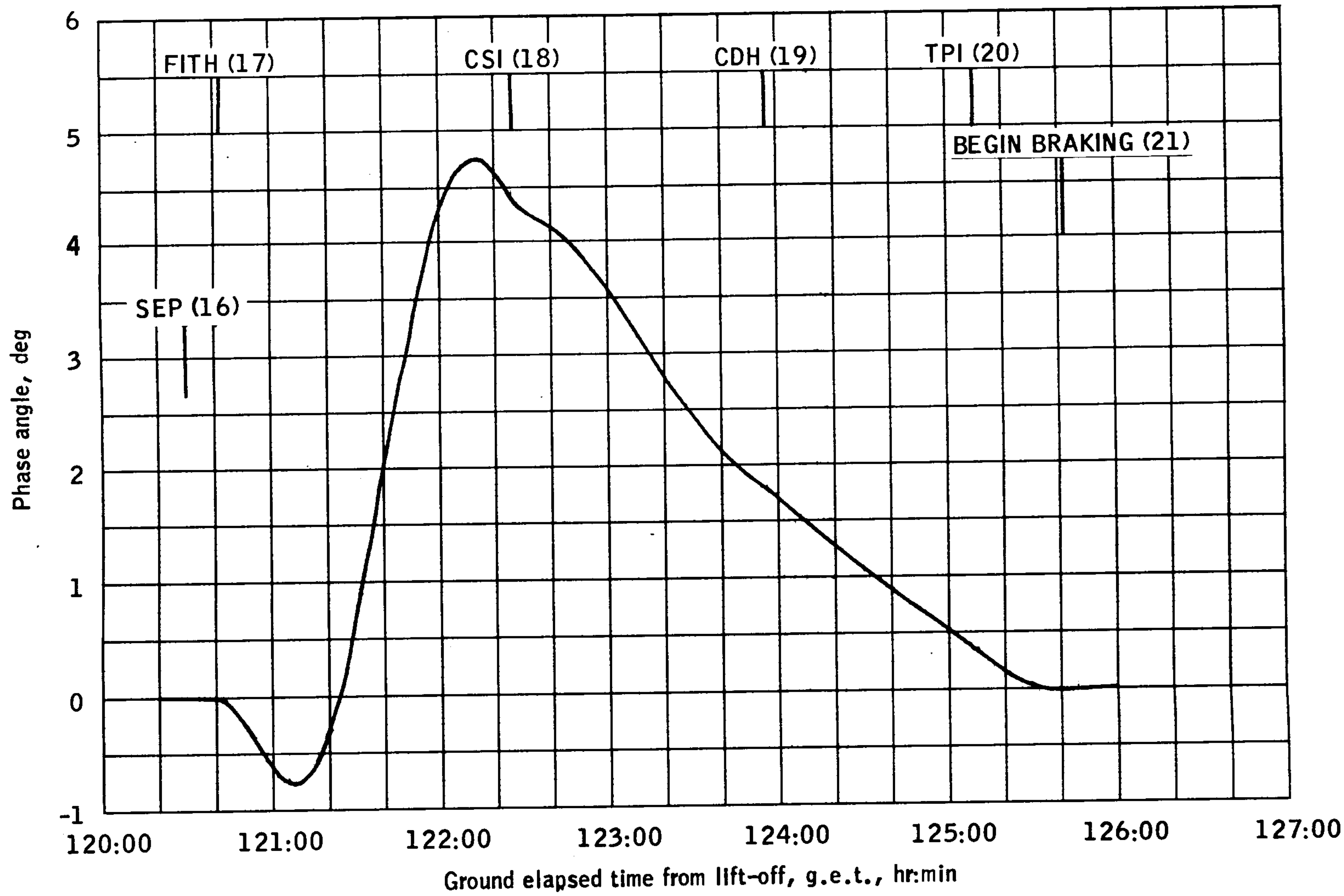


Figure 18.- Time history of phase angle between LM and CSM during LM-active rendezvous.

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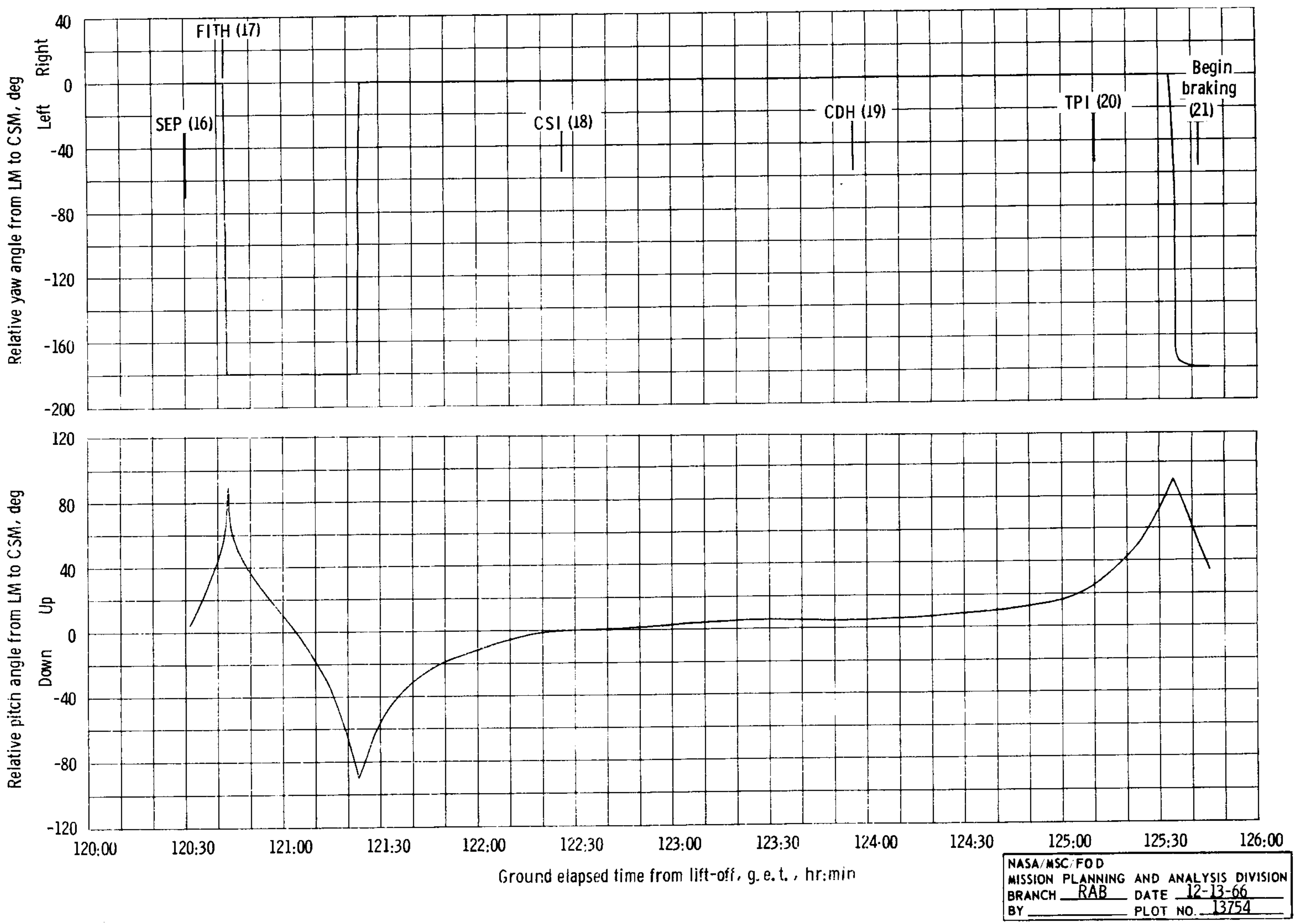

Figure 19. - Time history of LM yaw and pitch angles to the CSM during LM-active rendezvous.

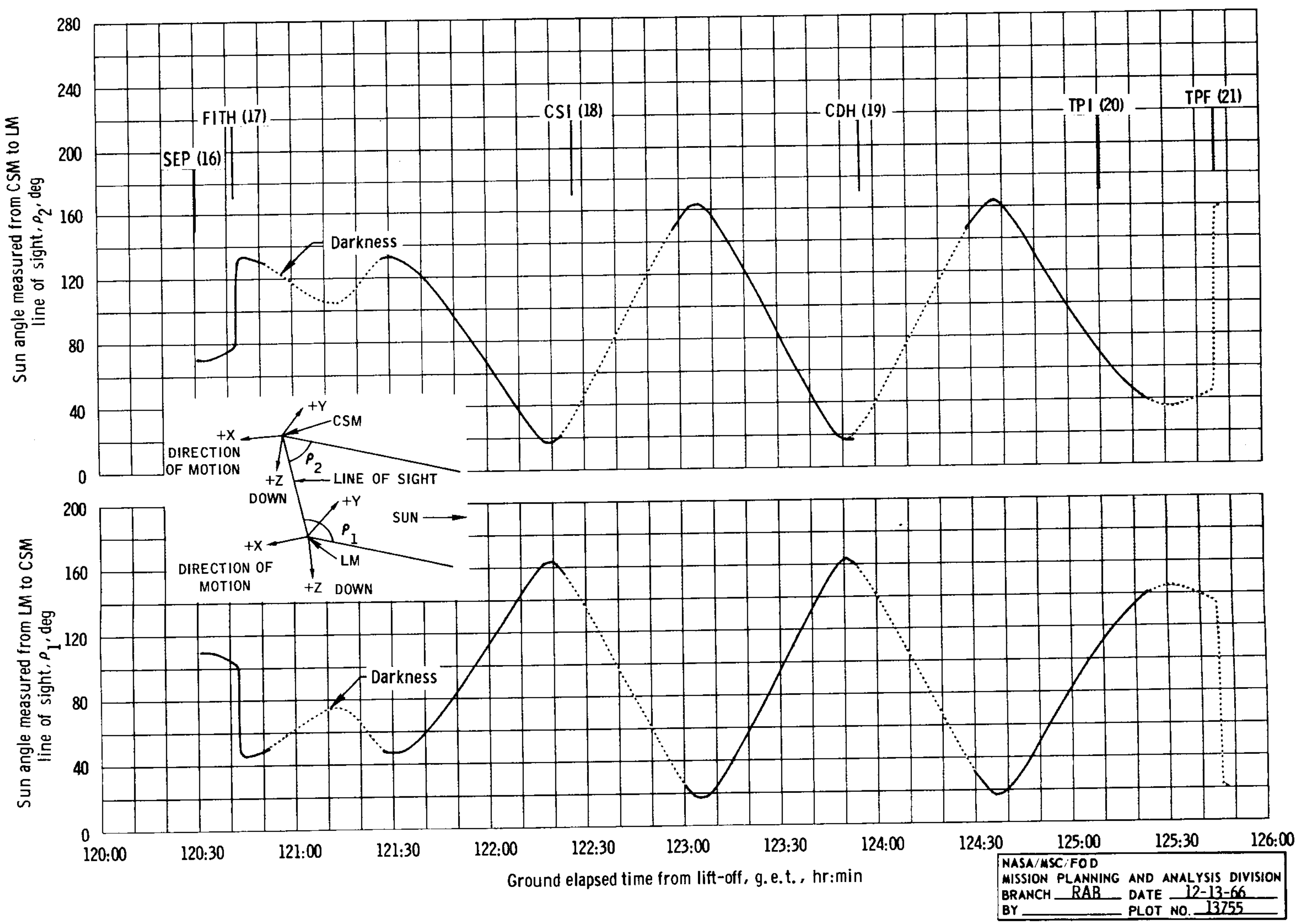

Figure 20. - Time history of angles to the sun measured from LM to CSM and from CSM to LM line of sight during LM-active rendezvous.

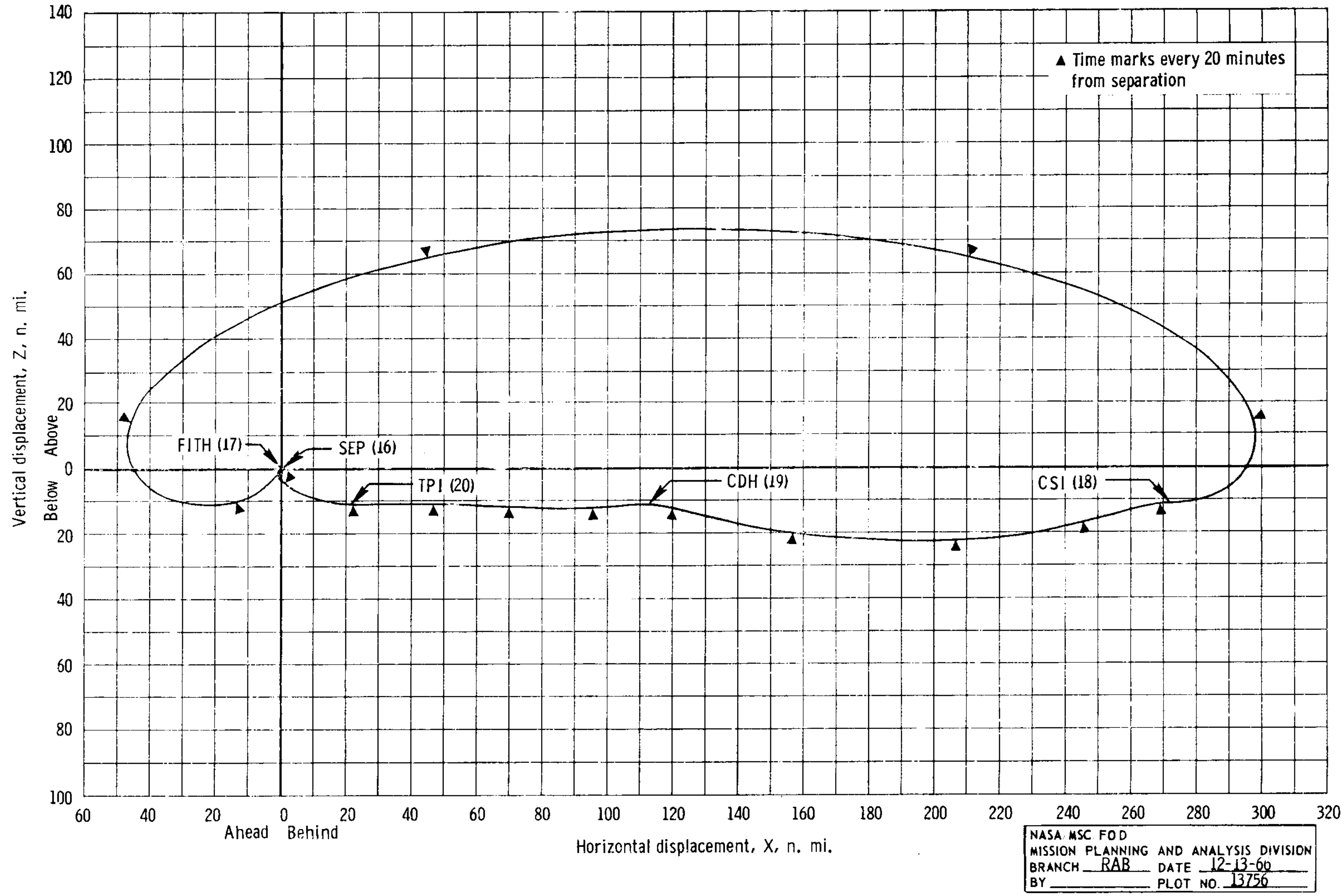

Figure 21. - Relative trajectory of LM from separation to rendezvous in CSM curvilinear coordinate system.

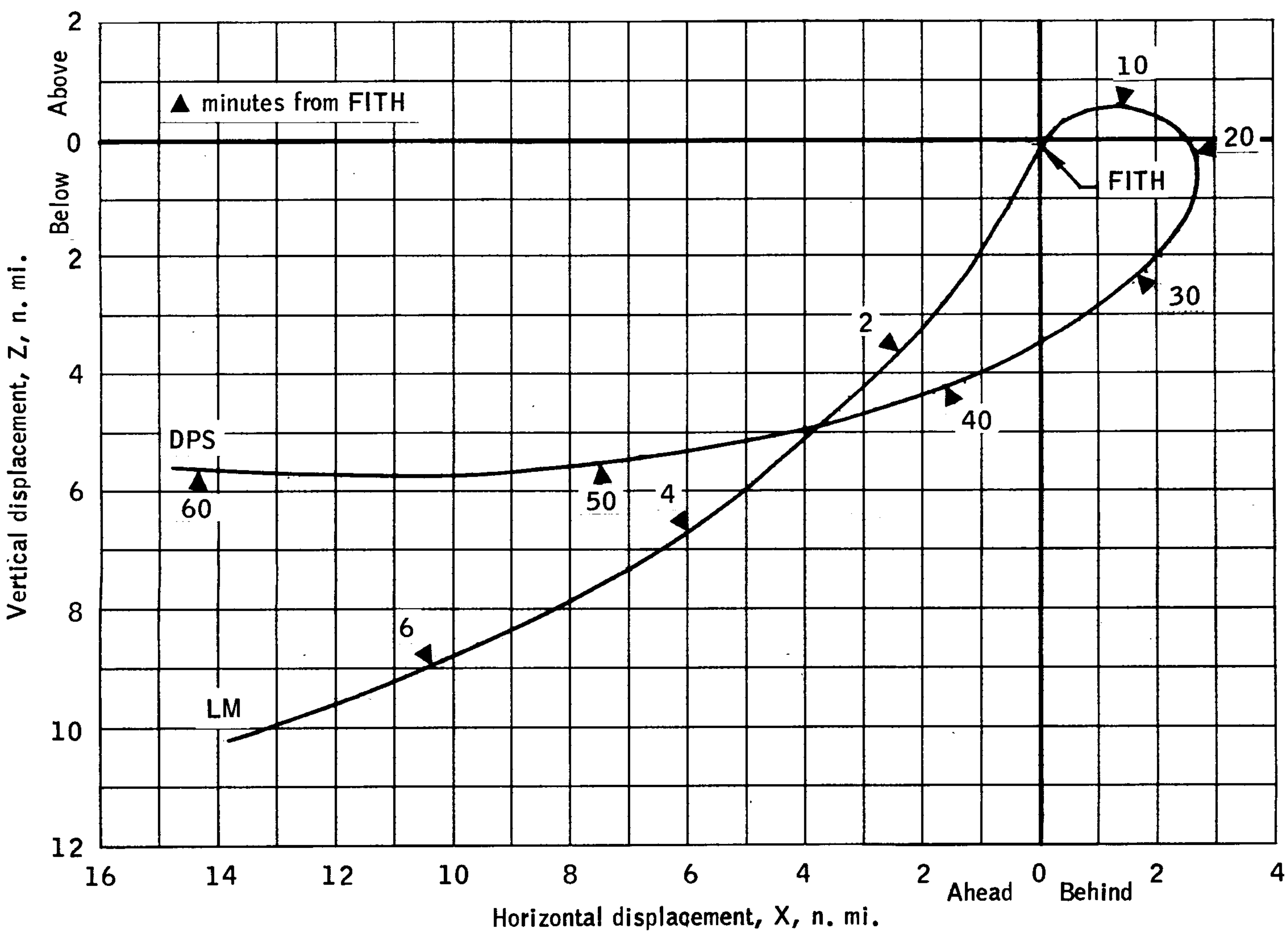

Figure 22.- Relative trajectories of LM and DPS during initial separation in CSM curvilinear coordinate system.

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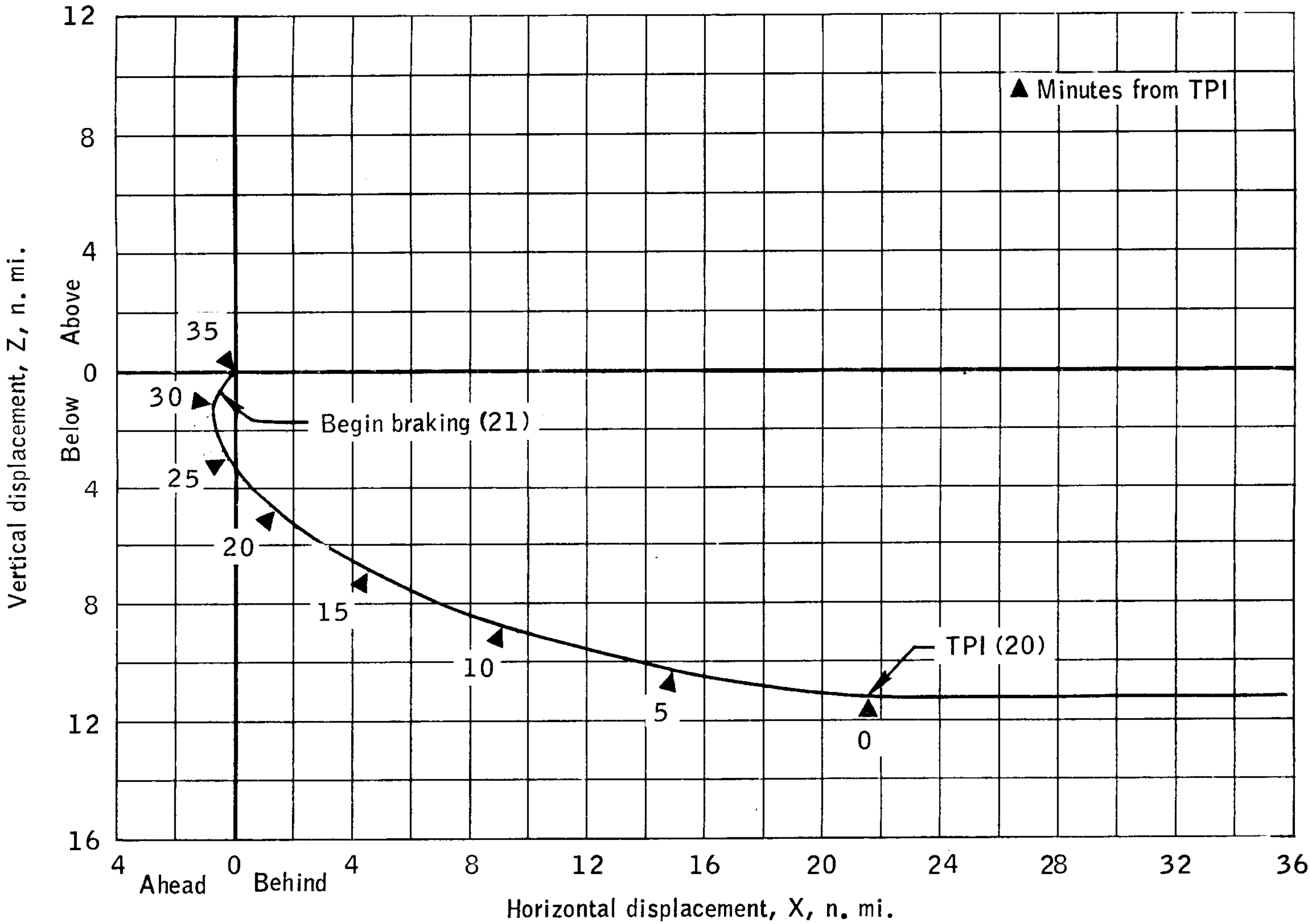

Figure 23.- Relative trajectory of LM from TPI to rendezvous in CSM curvilinear coordinate system.

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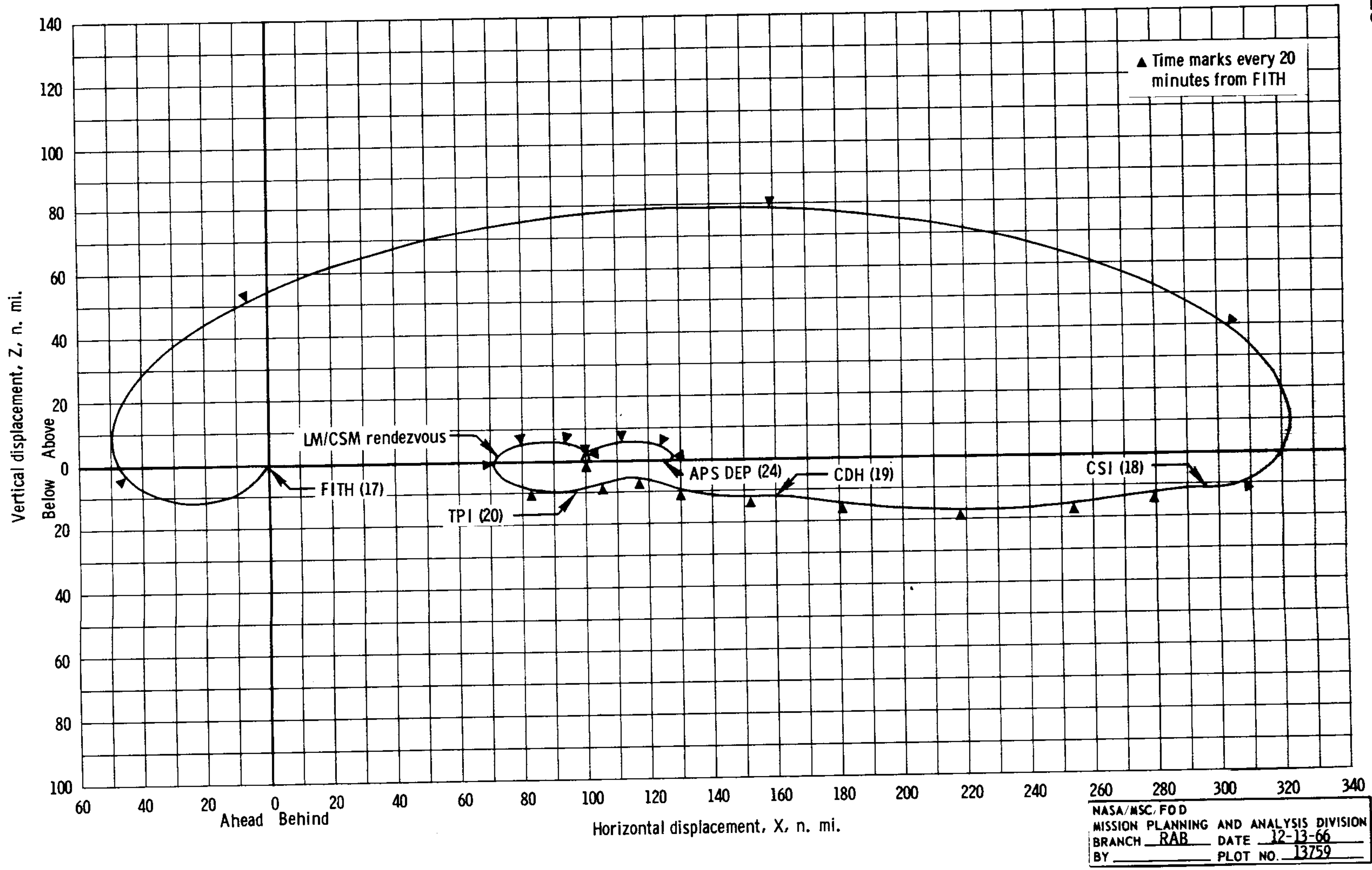


Figure 24. - Relative trajectory of LM to DPS from FITH to time of APS burn to fuel depletion in DPS curvilinear coordinate system.

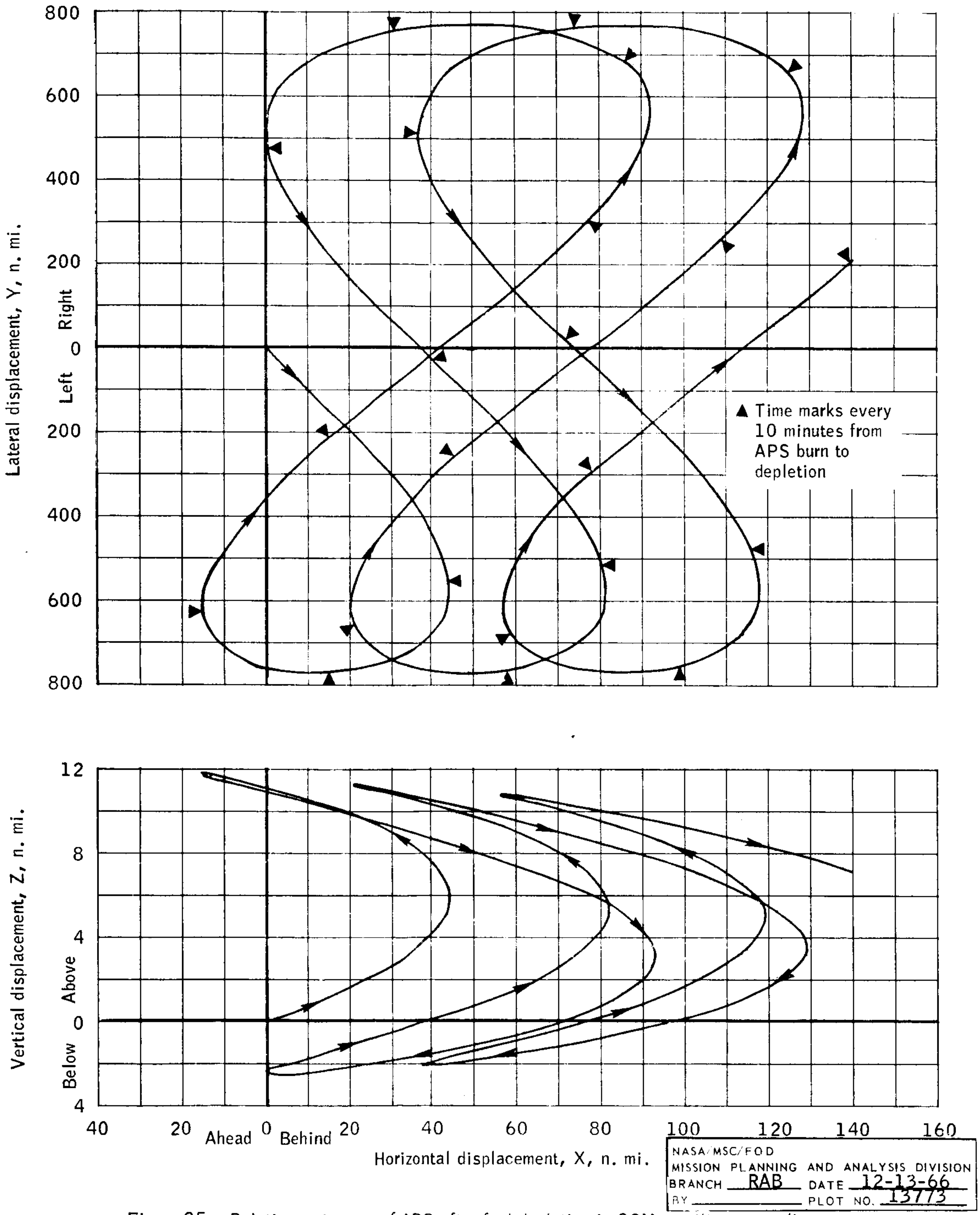
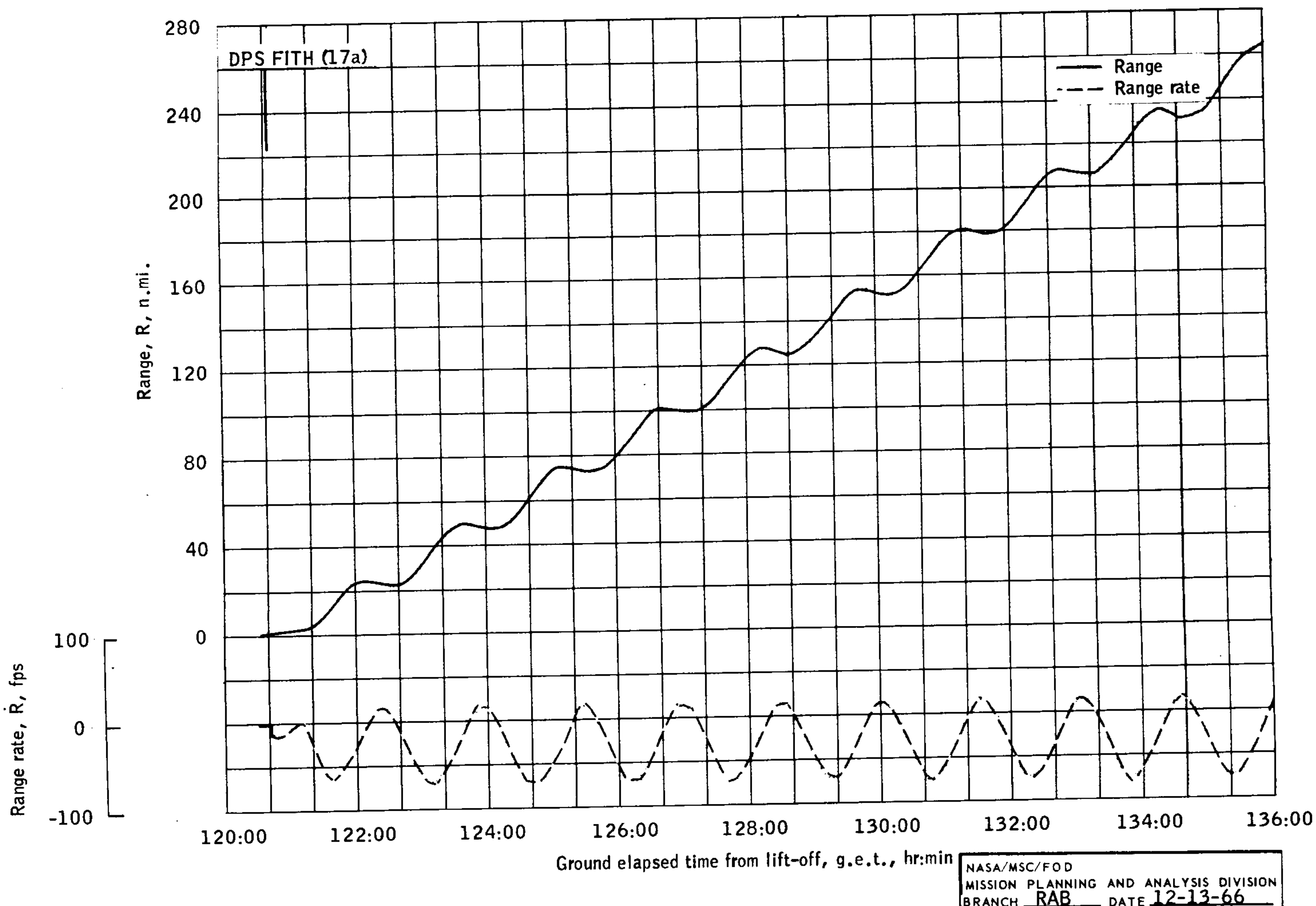
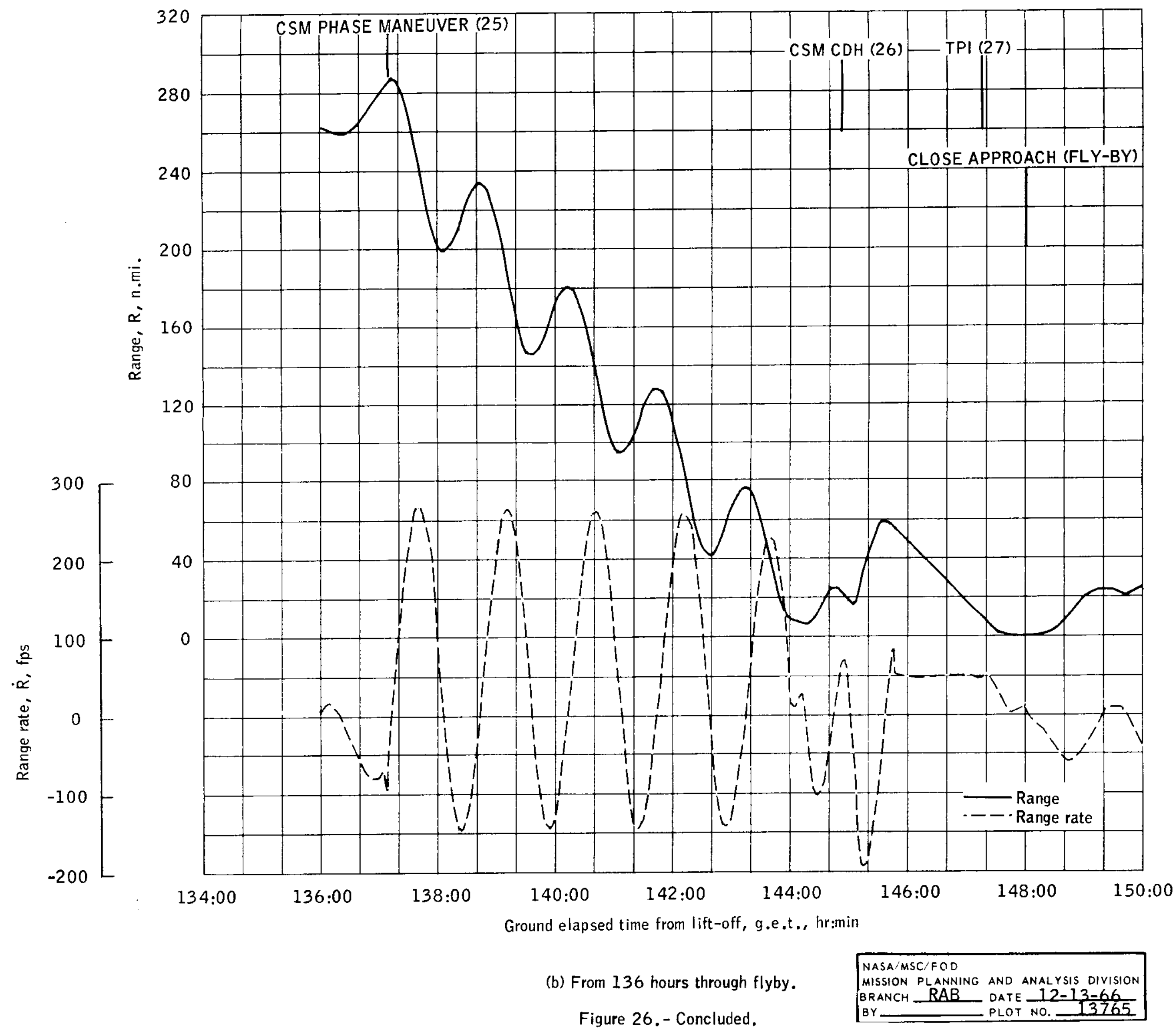


Figure 25.- Relative trajectory of APS after fuel depletion in CSM curvilinear coordinate system.



(a) From FITH to 136 hours.

Figure 26.- Time history of range and range rate between CSM and DPS during CSM-active flyby.



(b) From 136 hours through flyby.

Figure 26.- Concluded.

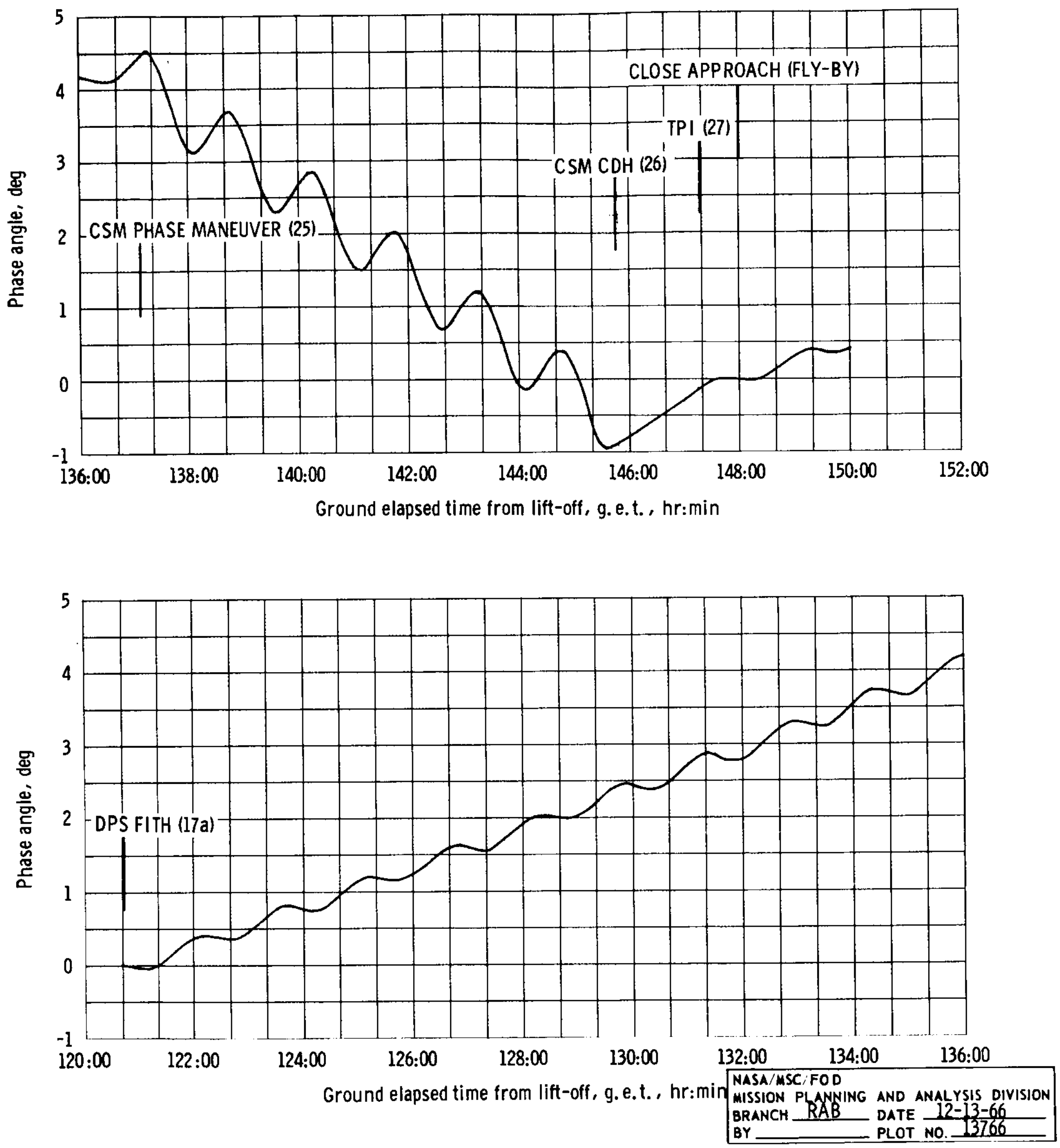
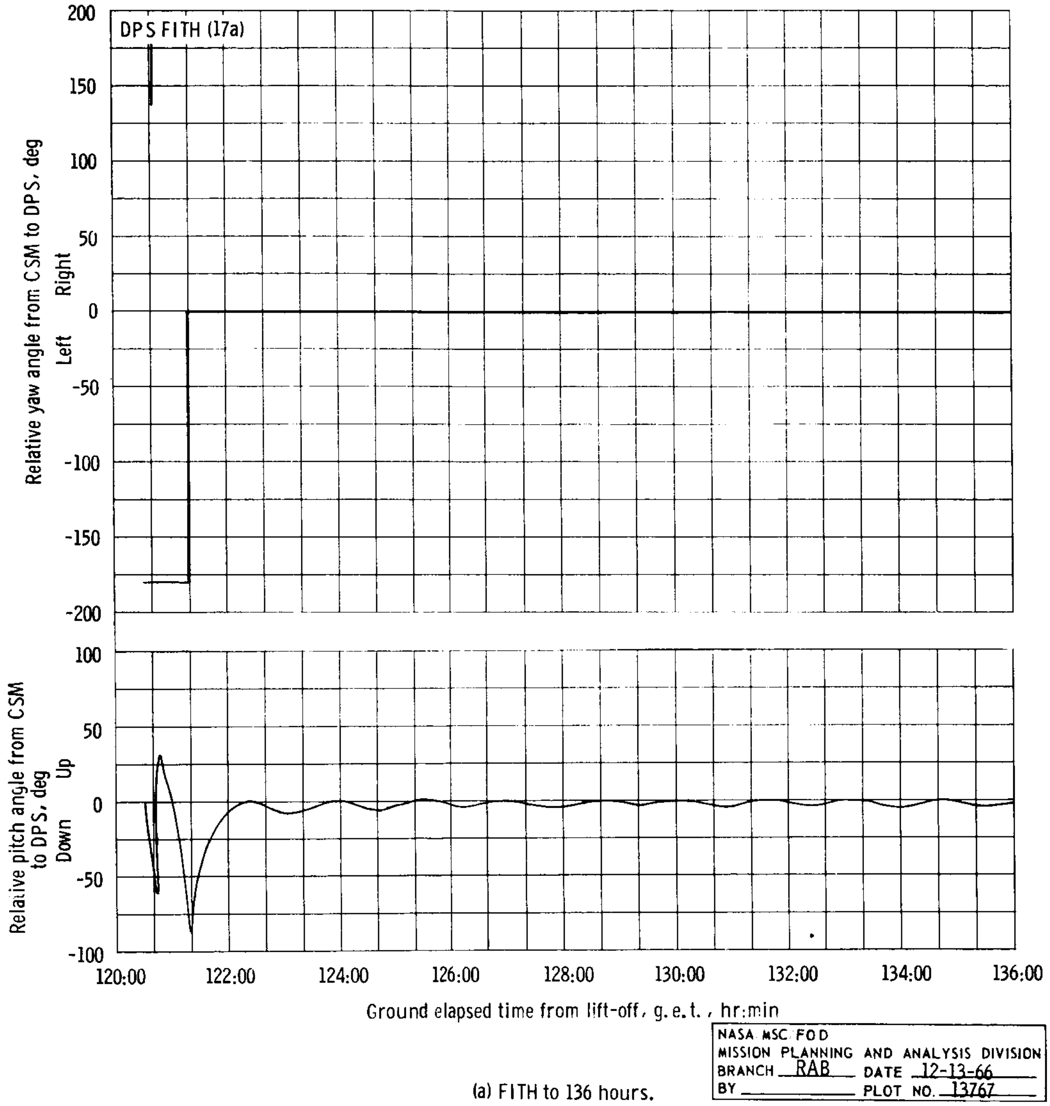
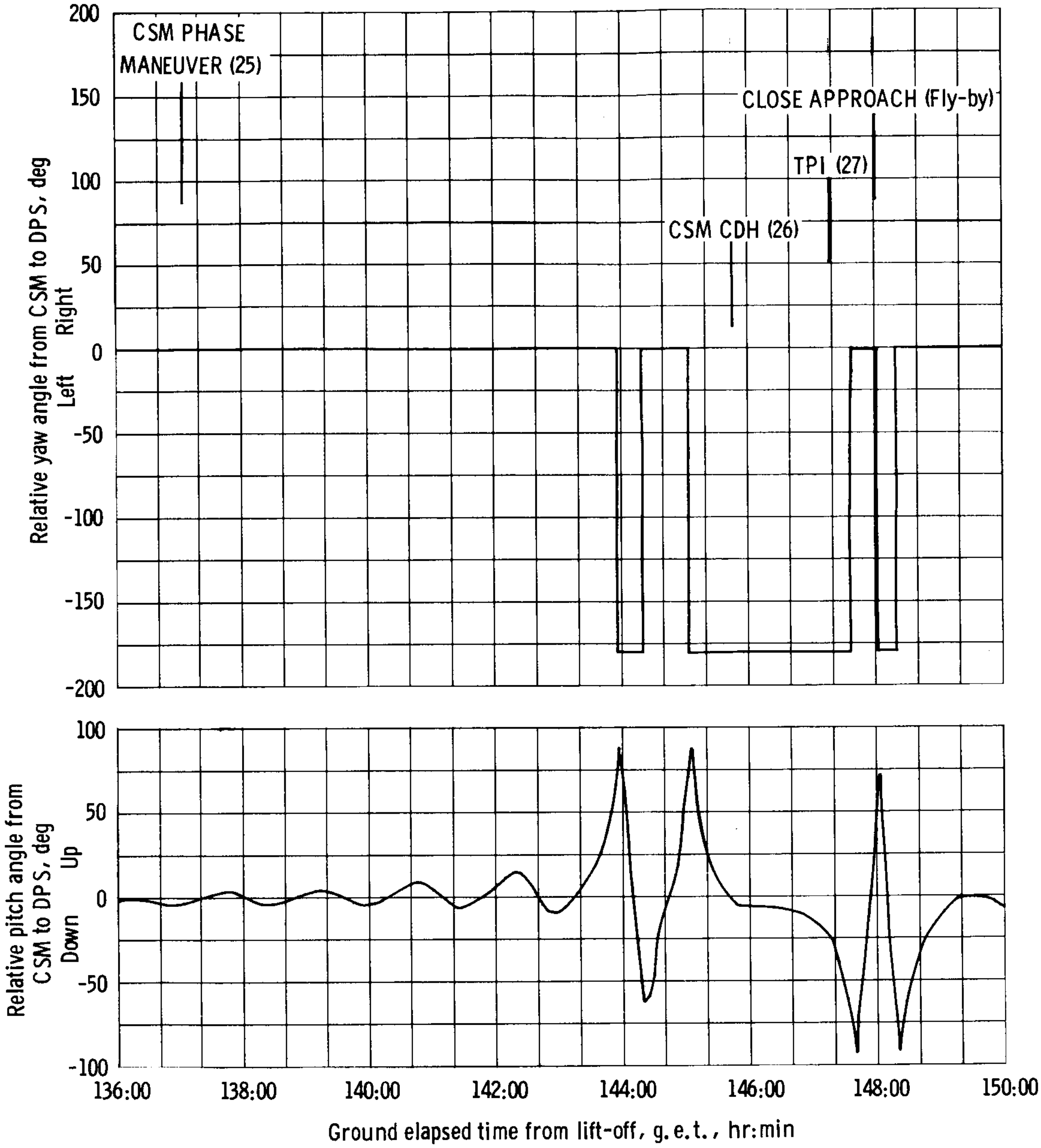


Figure 27.- Time history of phase angle between CSM and DPS from FITH through flyby.



(a) FITH to 136 hours.

Figure 28. - Time history of CSM yaw and pitch angles to the DPS during CSM-active flyby.



(b) 136 hours through flyby.

Figure 28. - Concluded.

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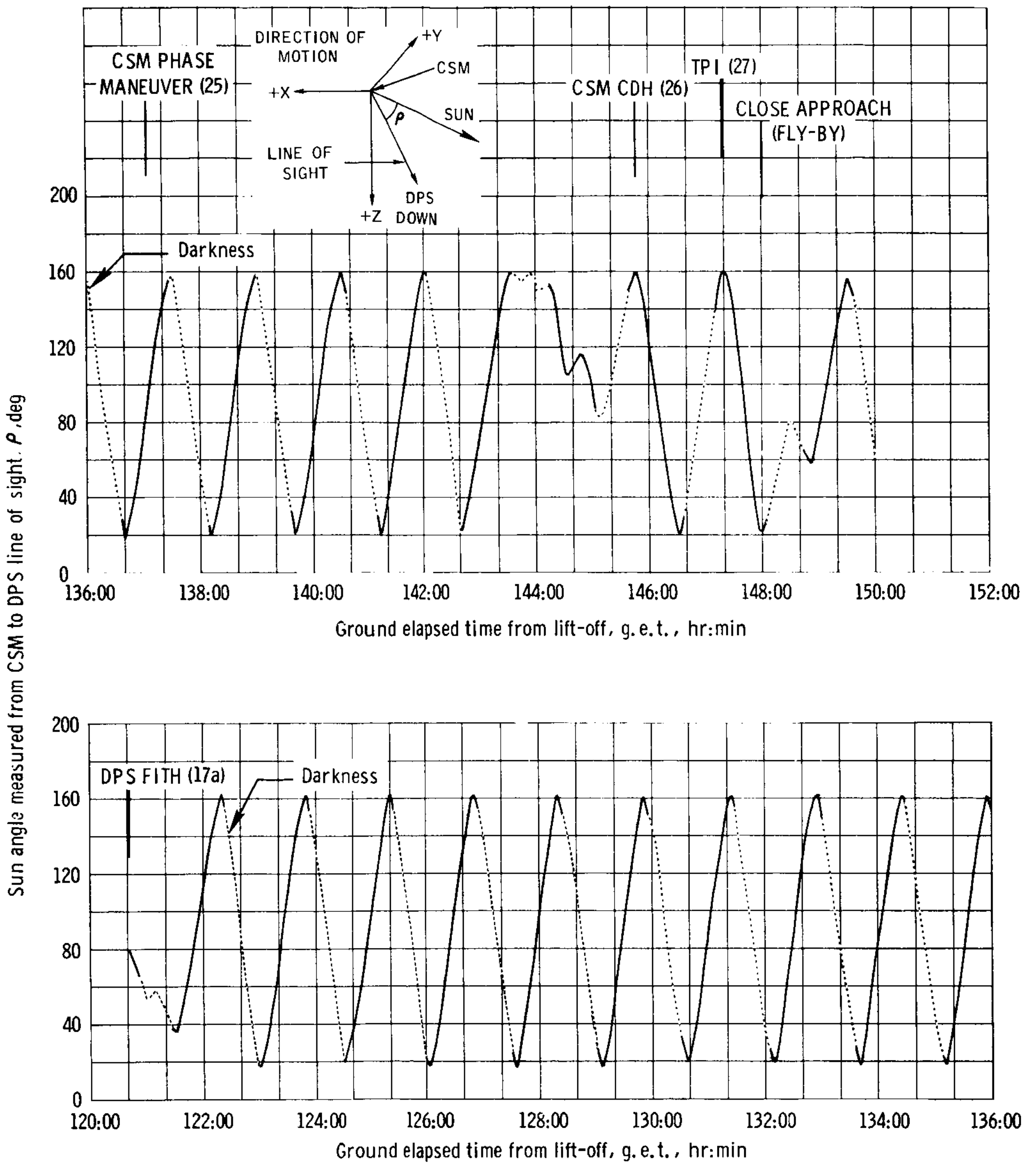

Figure 29. - Time history of sun angle (measured from CSM-DPS line of sight) during CSM active flyby.

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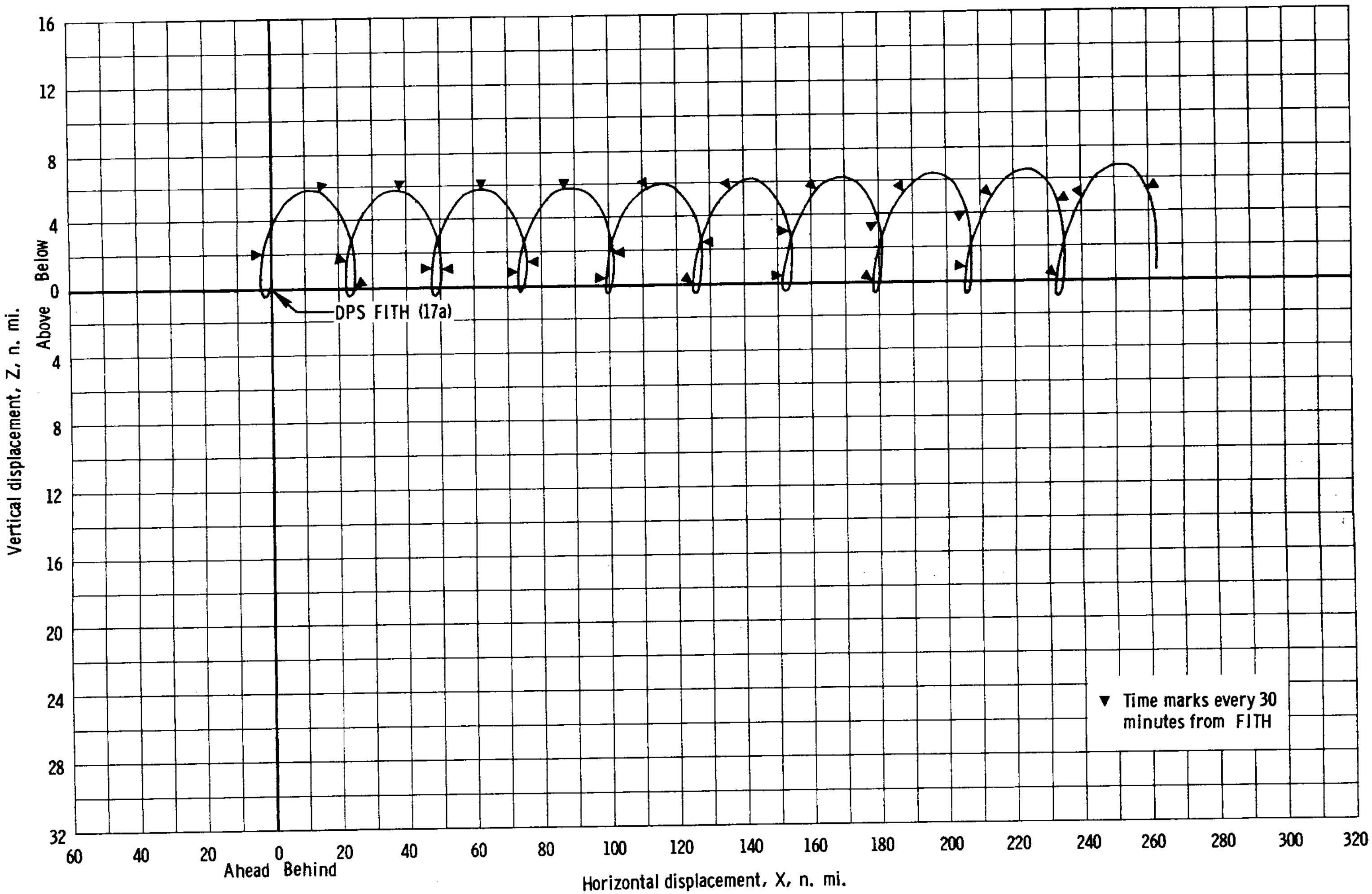

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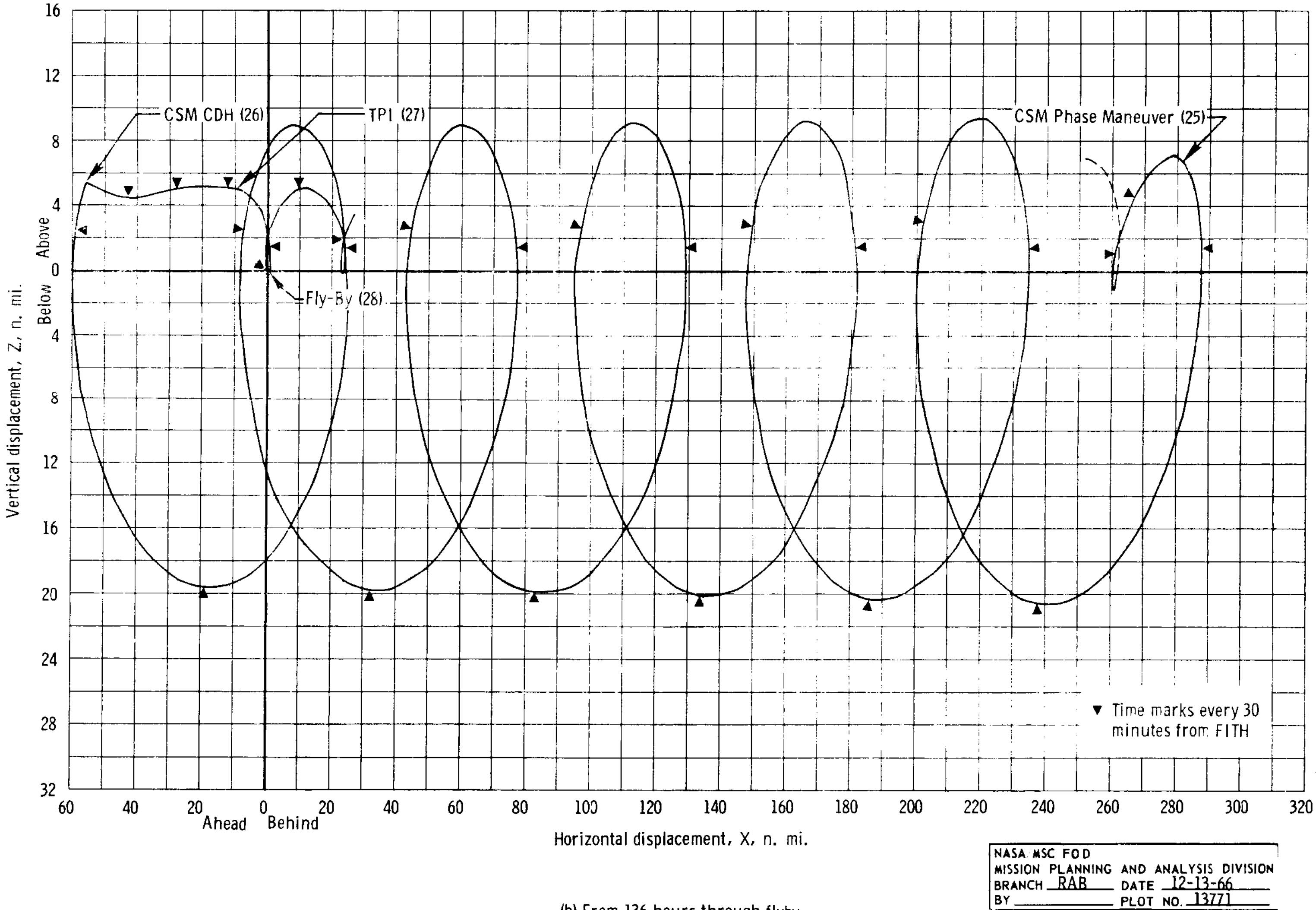
(a) From FITH to 136 hours.

Figure 30. - Relative trajectory of CSM to DPS during CSM-active flyby in DPS curvilinear coordinate system.



(b) From 136 hours through flyby.

Figure 30. - Concluded.

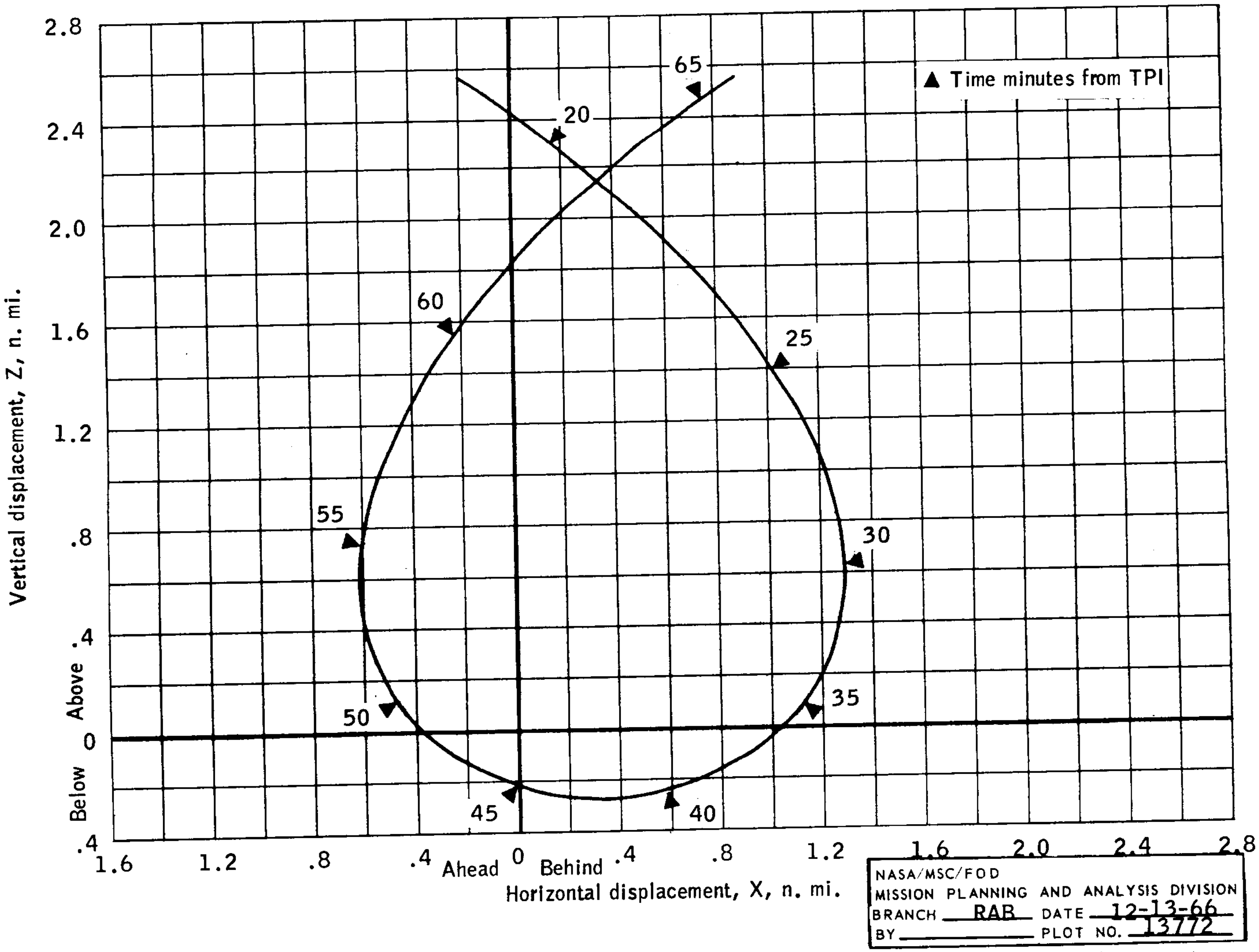
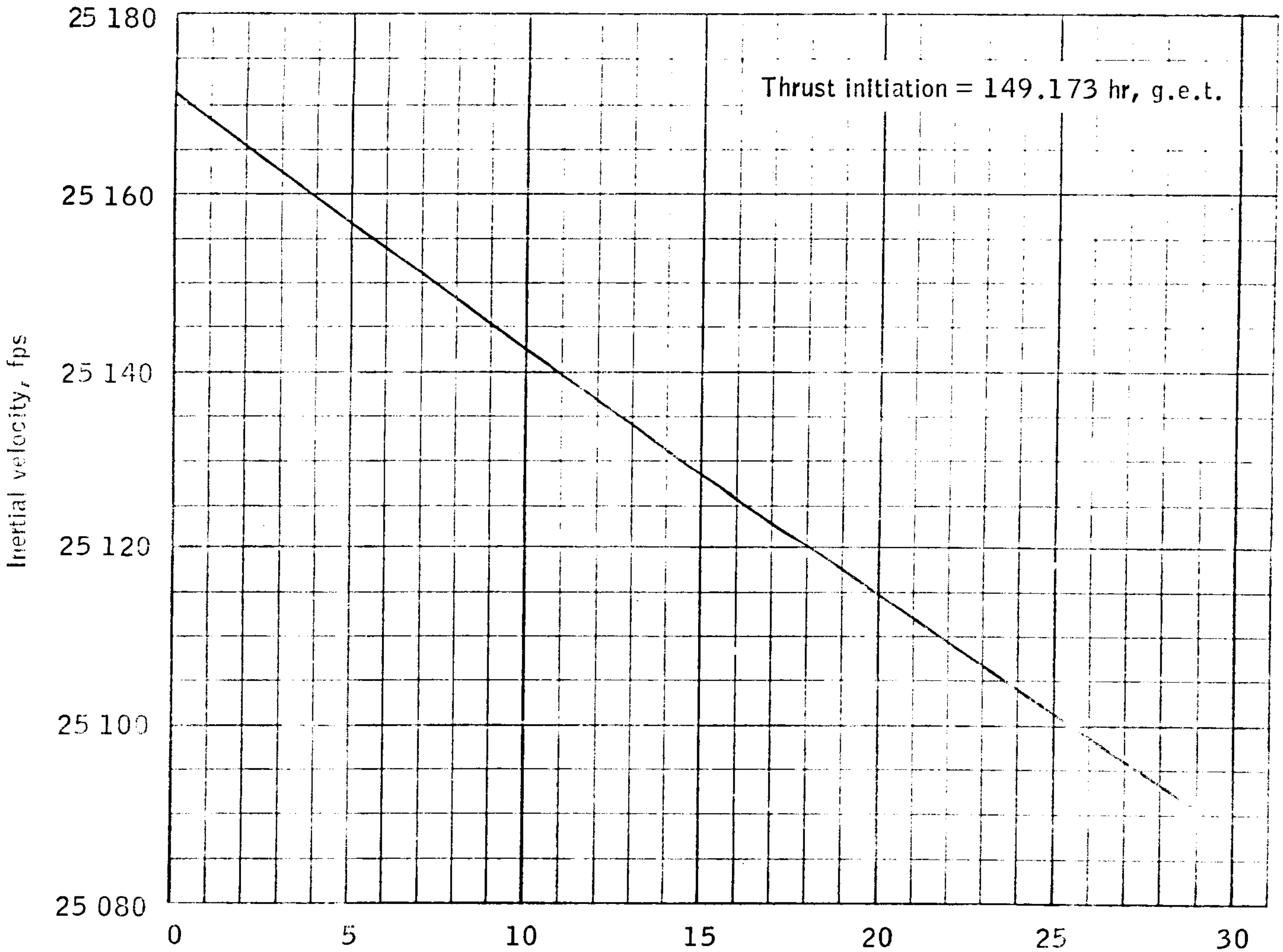
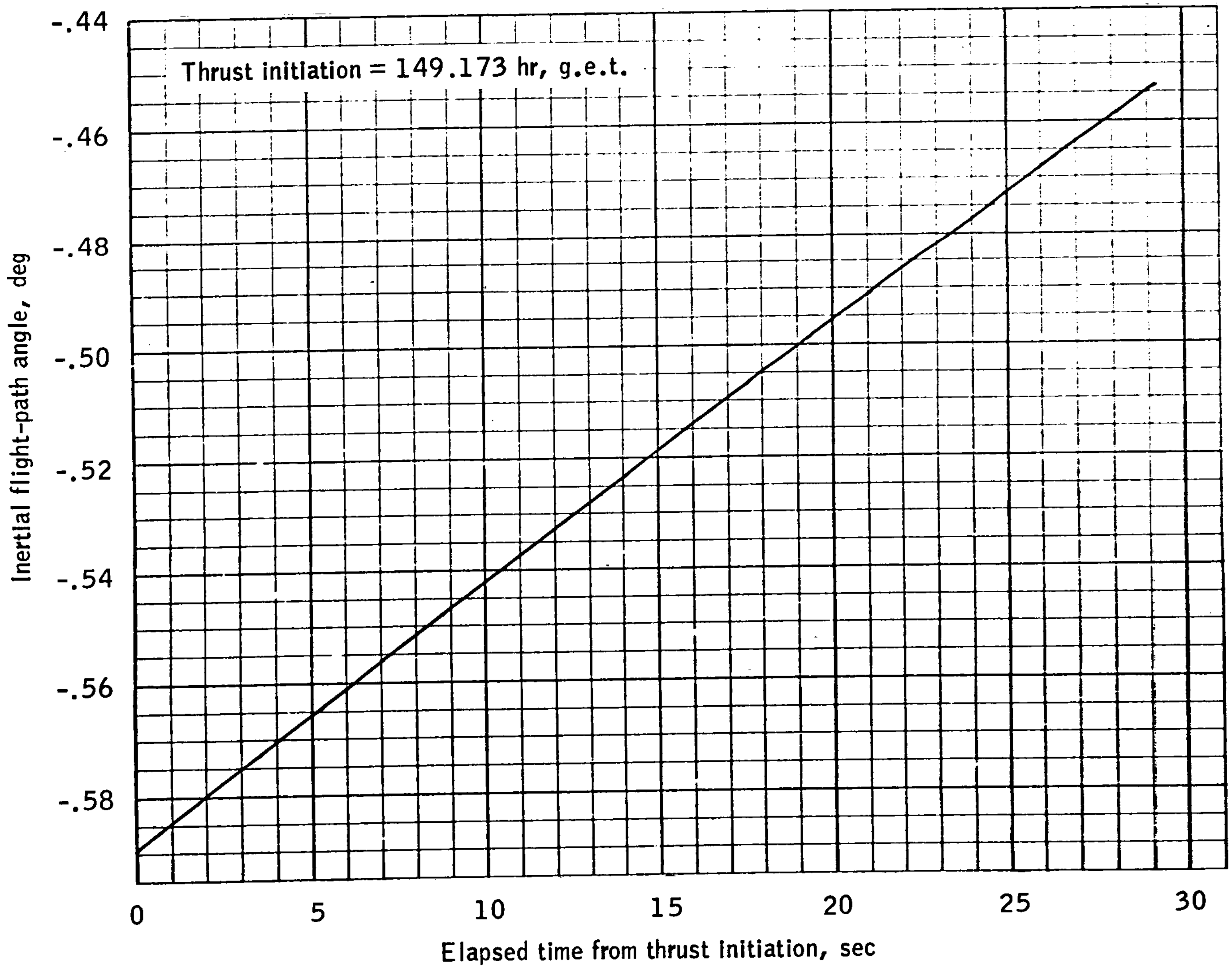


Figure 31. - Relative trajectory of CSM during time of close approach to DPS in DPS curvilinear coordinate system.



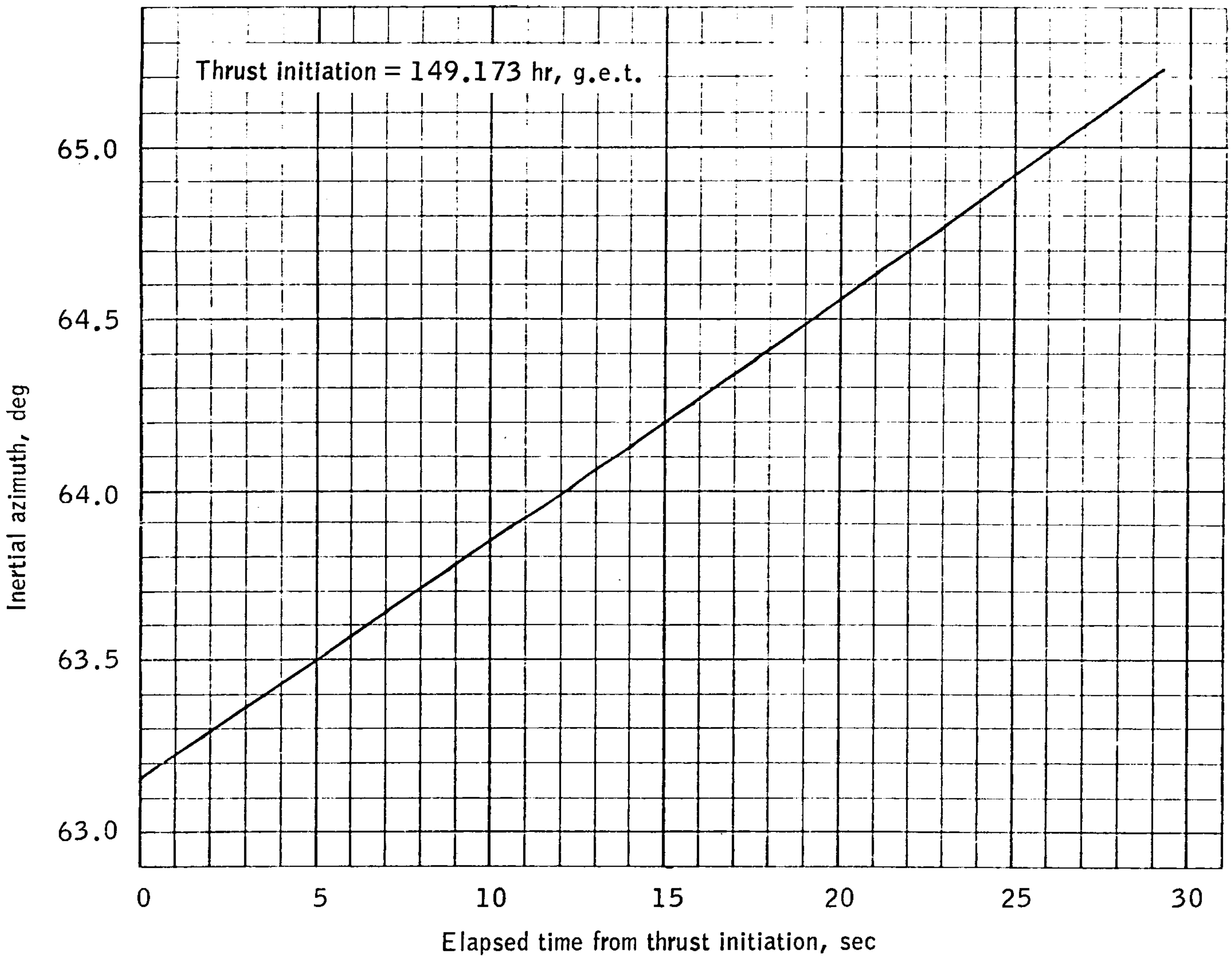
(a) Inertial velocity versus time from thrust initiation.

Figure 32. - Trajectory parameter time history for simulated transearth burn (event 29).



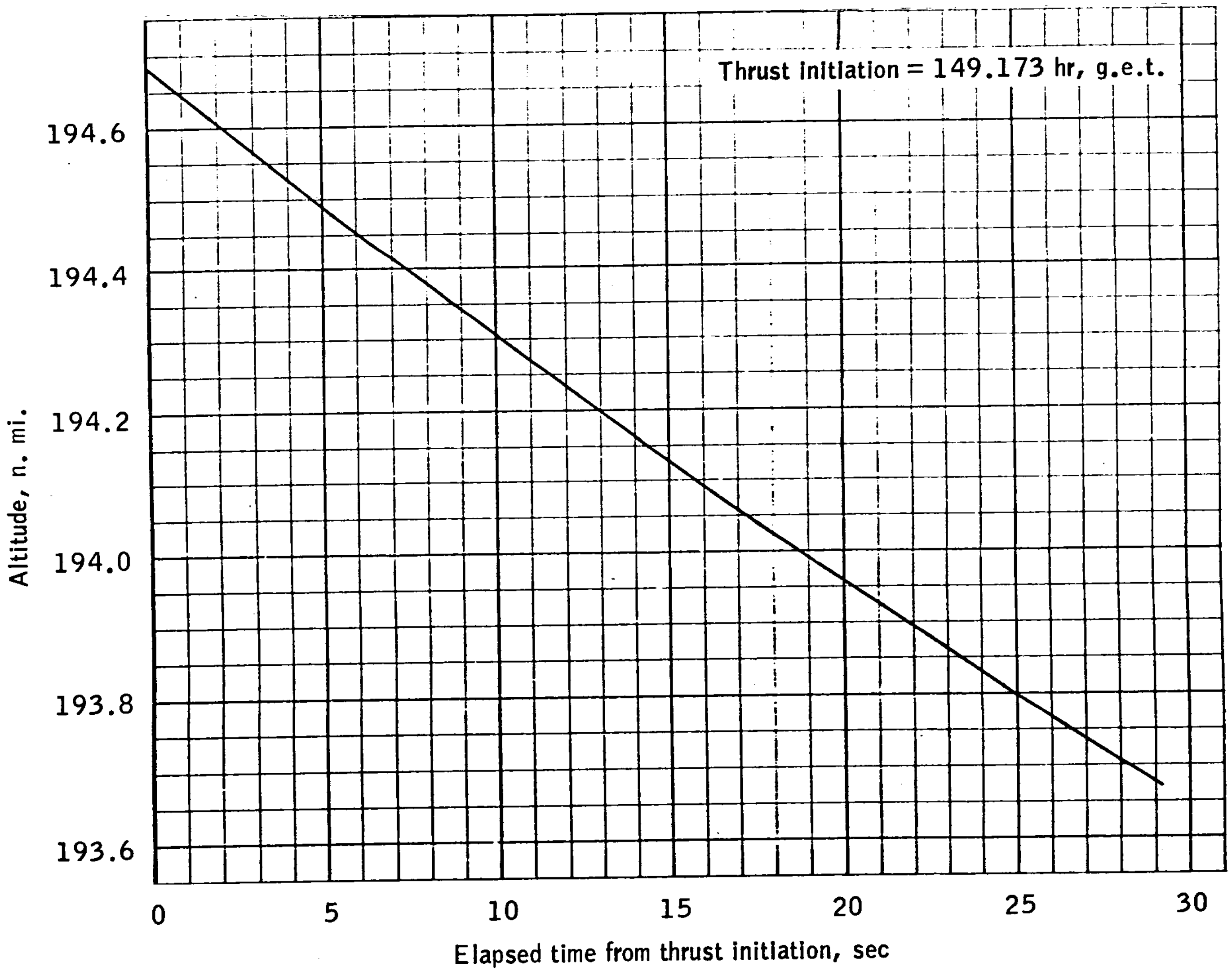
(b) Inertial flight-path angle versus time from thrust initiation.

Figure 32.- Continued.



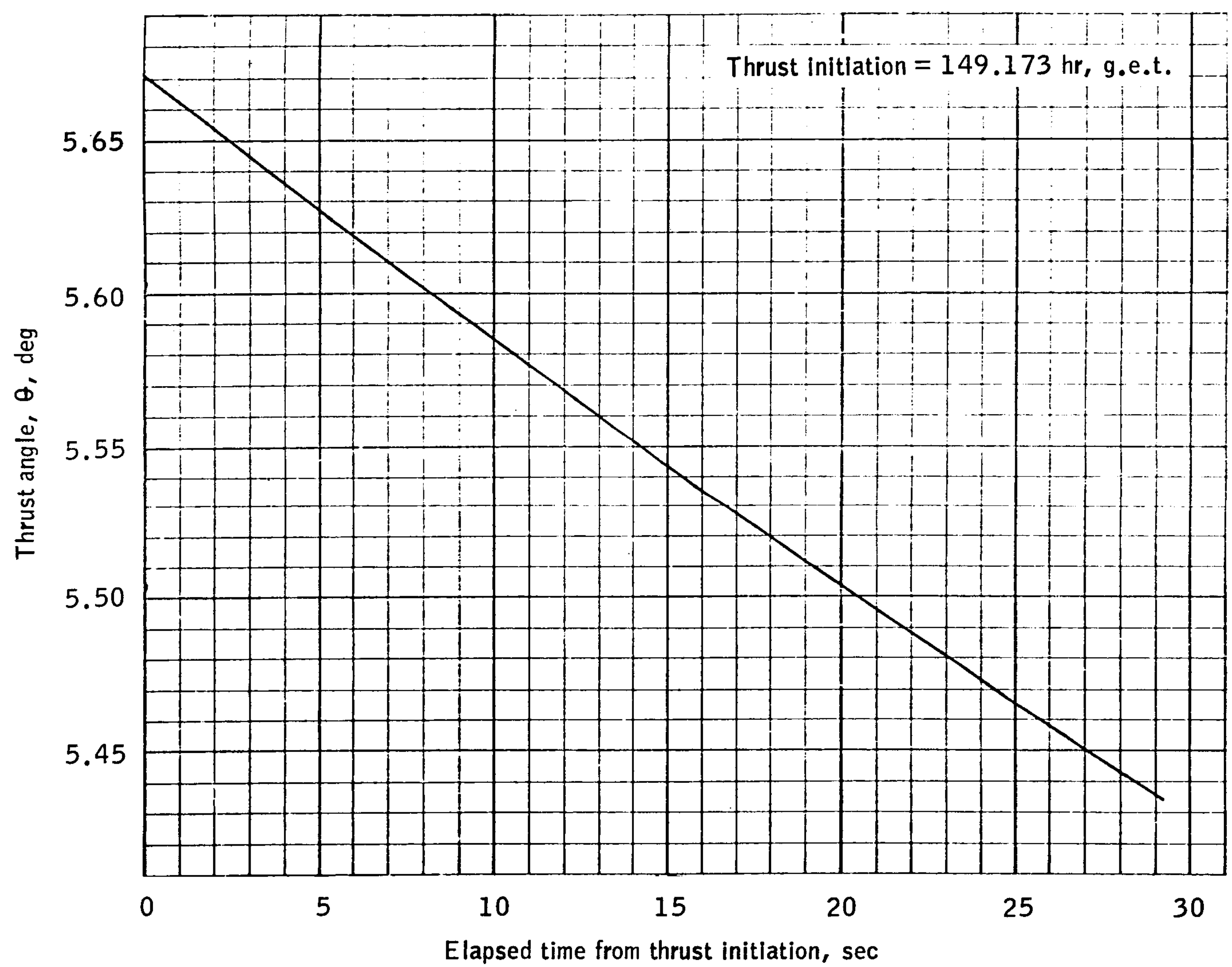
(c) Inertial azimuth versus time from thrust initiation.

Figure 32.- Continued.



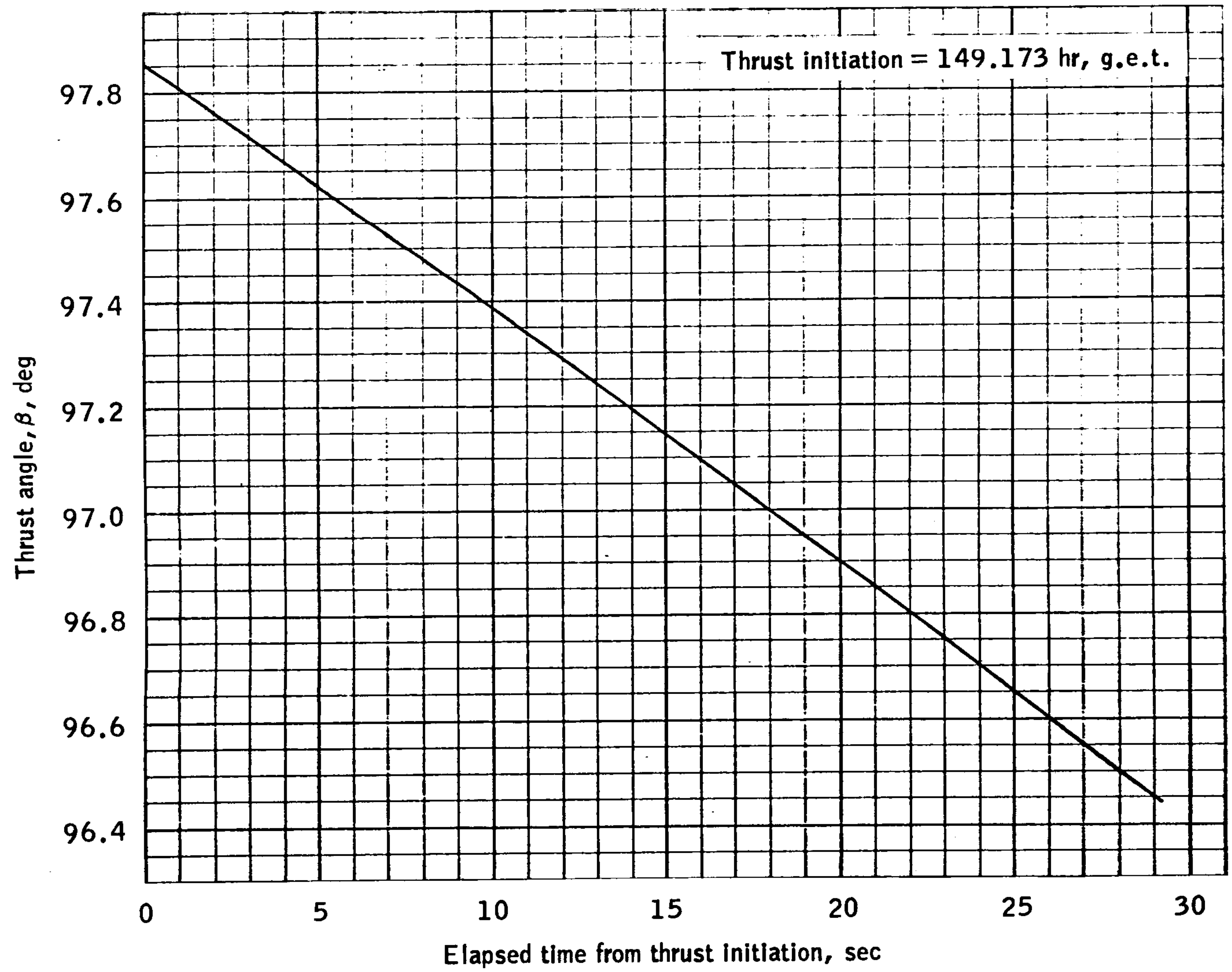
(d) Altitude versus time from thrust initiation.

Figure 32.- Continued.



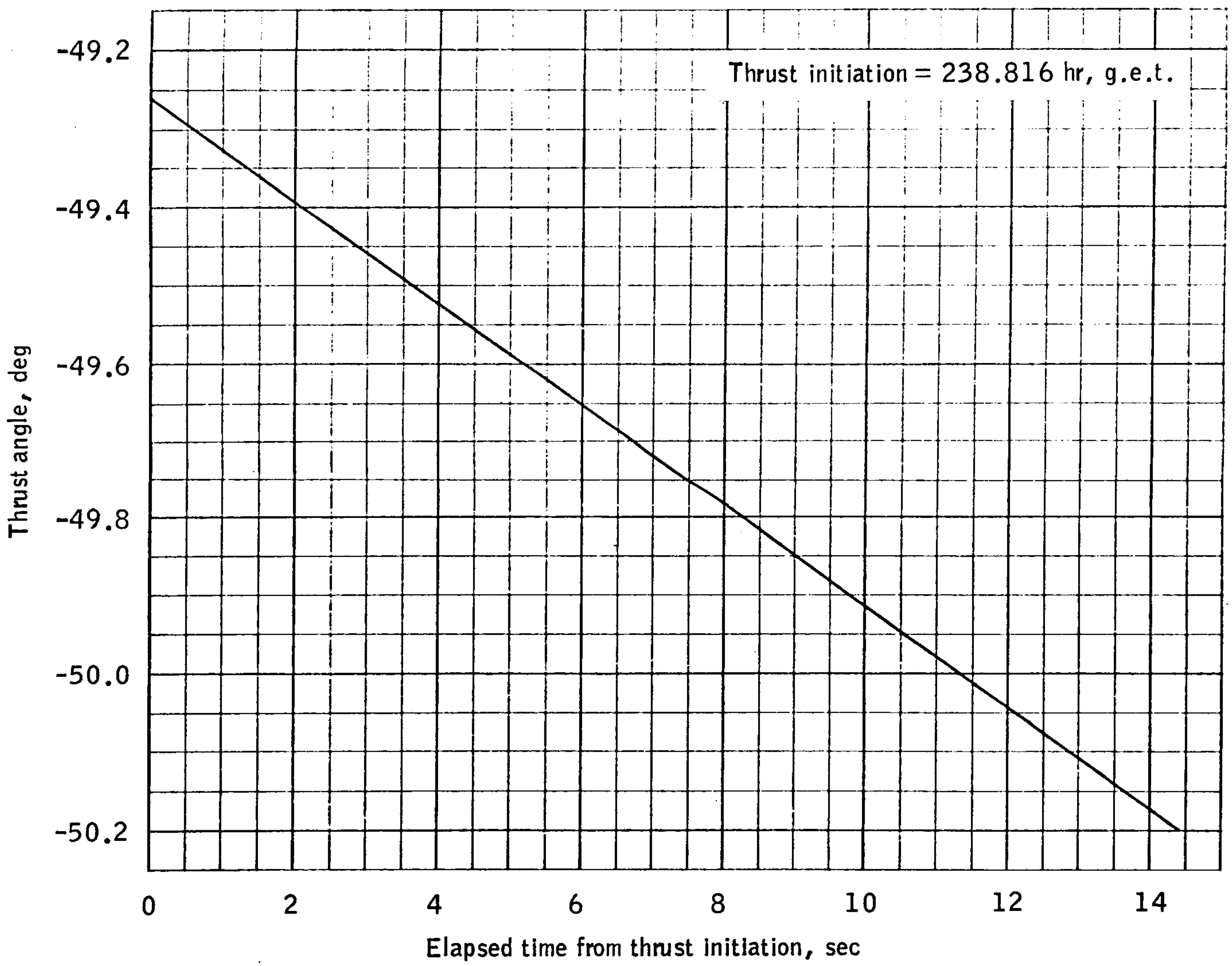
(e) Thrust angle theta versus time from thrust initiation.

Figure 32.- Continued.



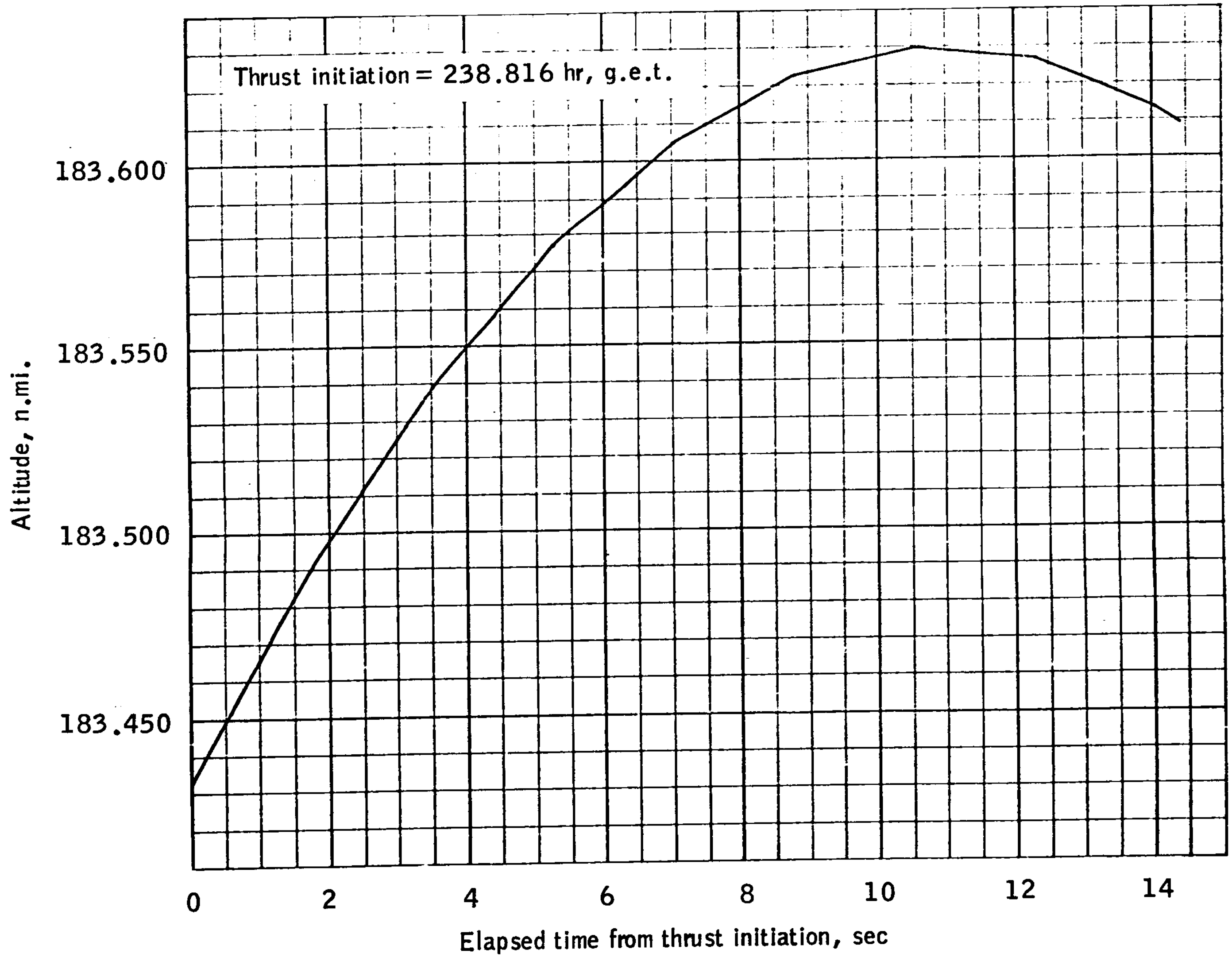
(f) Thrust angle beta versus time from thrust initiation.

Figure 32.- Concluded.



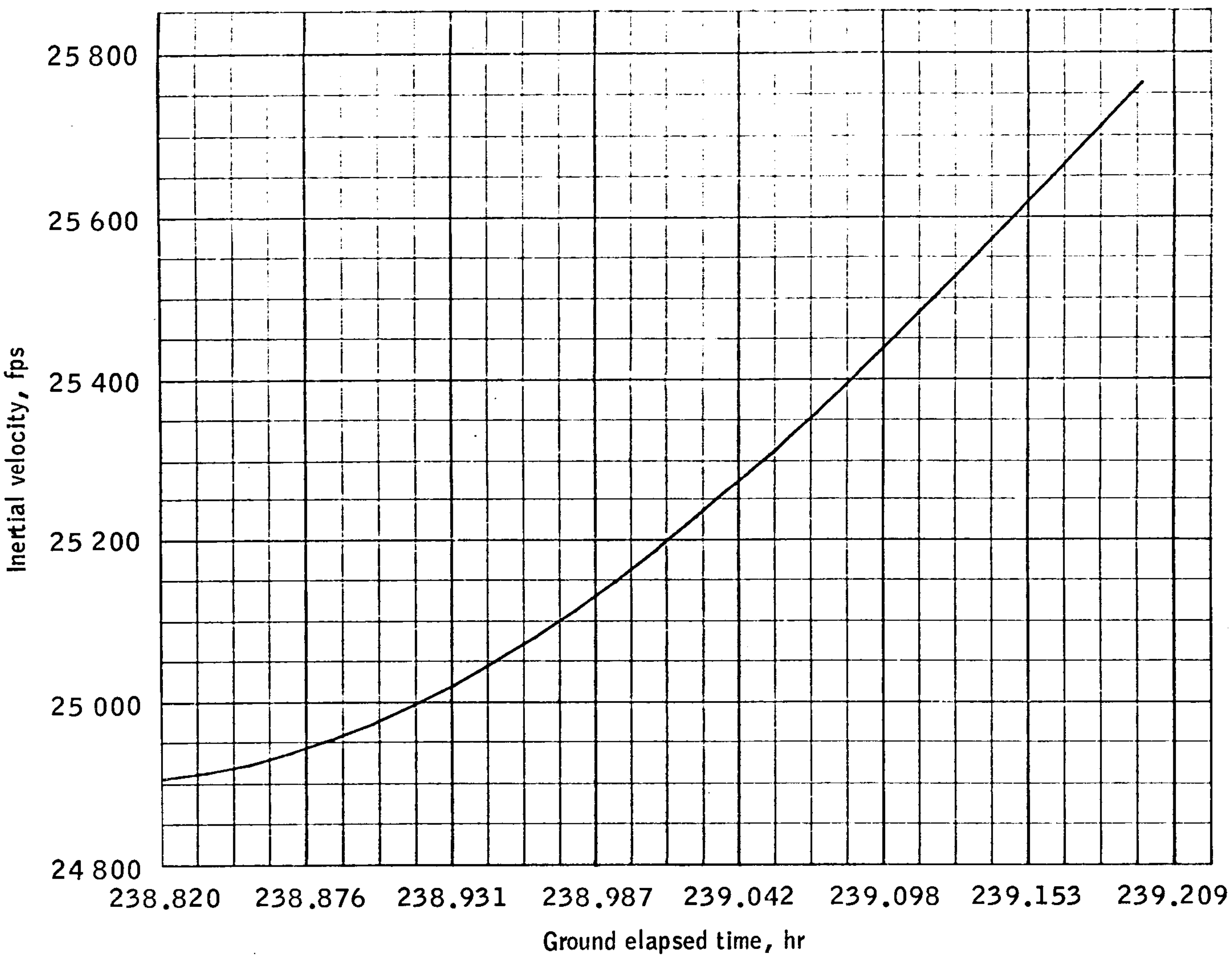
(a) Thrust angle theta versus time from thrust initiation.

Figure 33. - Trajectory parameter time history for deorbit burn (event 31).



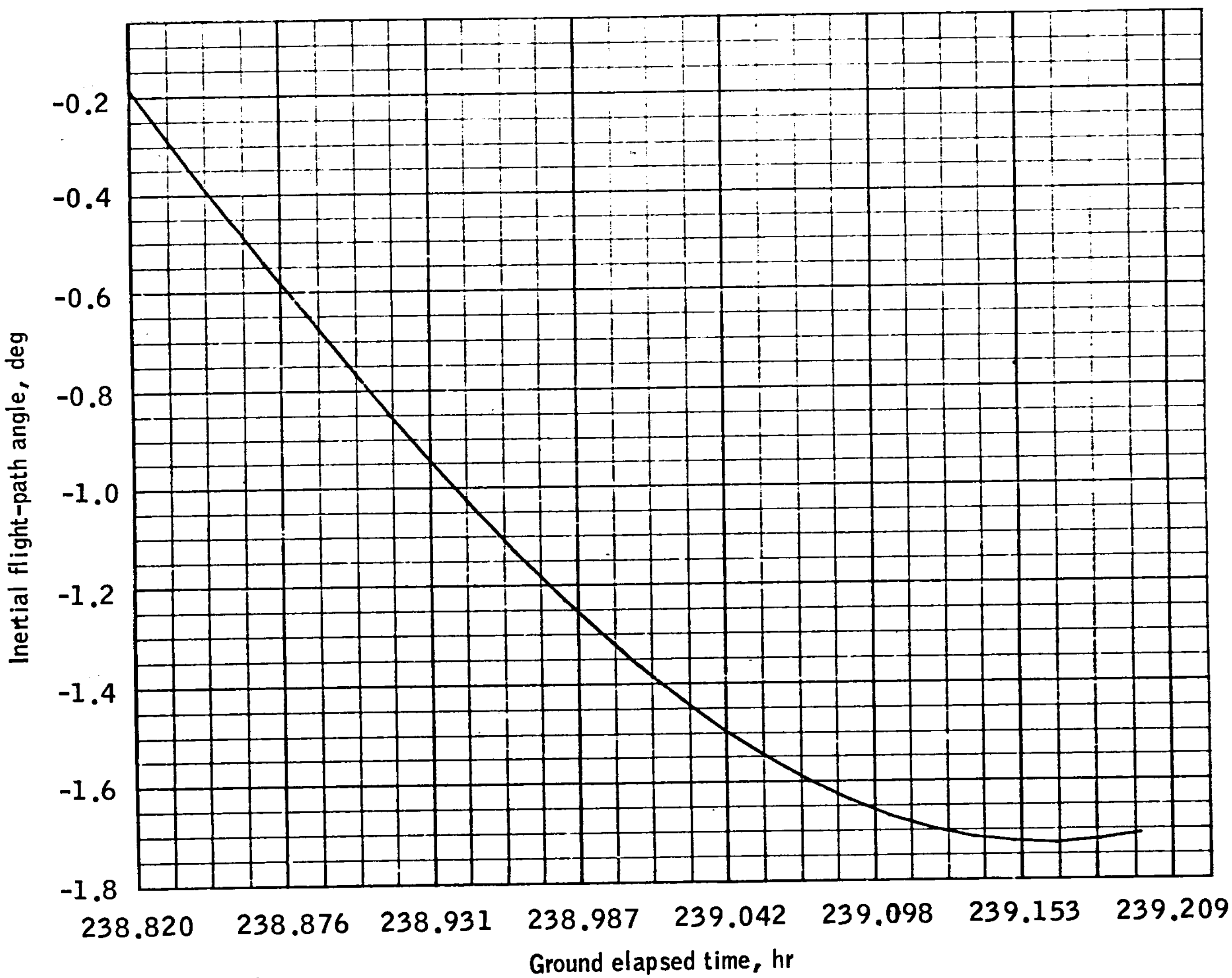
(b) Altitude versus time from thrust initiation.

Figure 33.- Concluded.



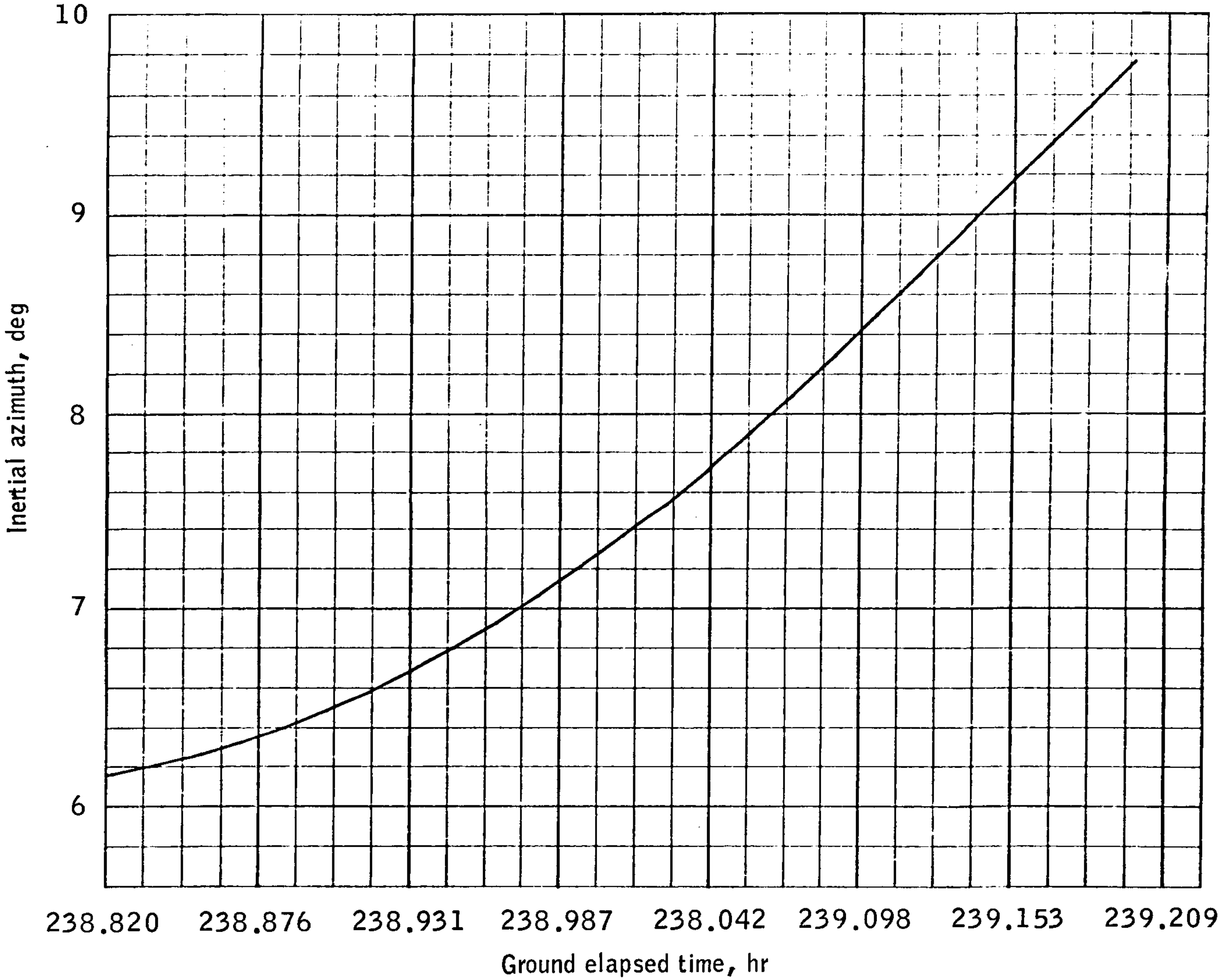
(a) Inertial velocity versus ground elapsed time.

Figure 34. - Trajectory parameter time history during coast to entry.



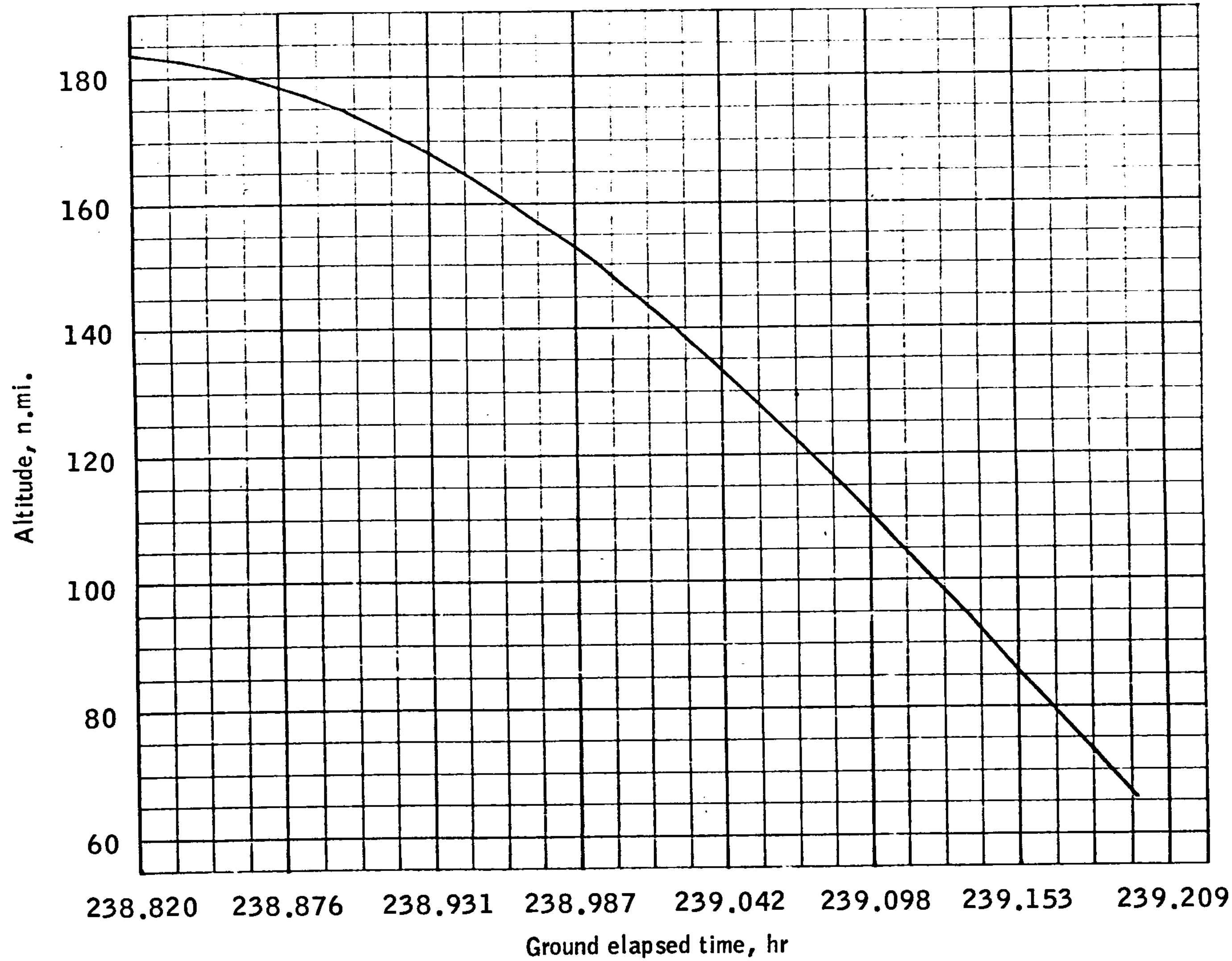
(b) Inertial flight-path angle versus ground elapsed time.

Figure 34.- Continued.



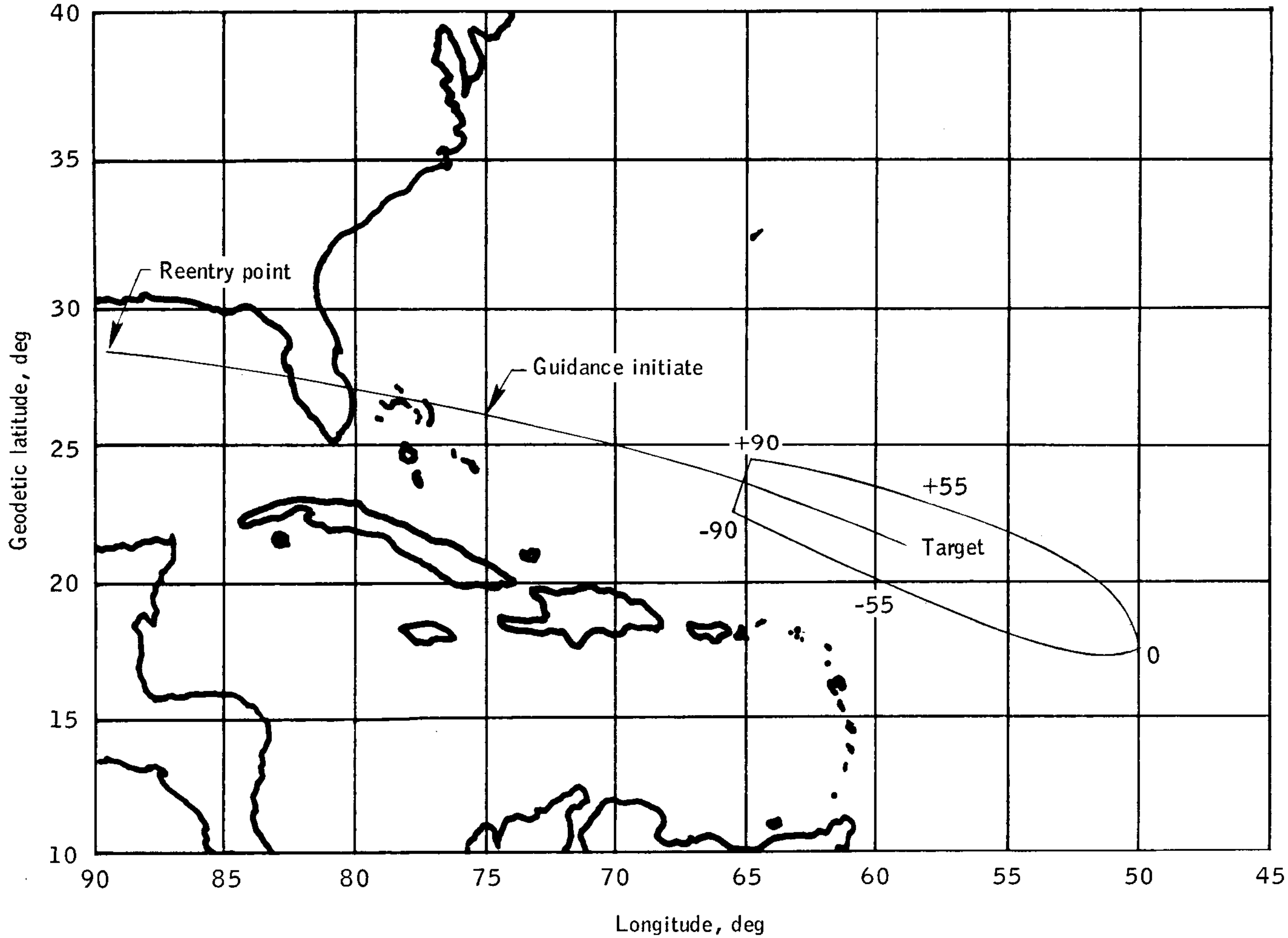
(c) Inertial azimuth versus ground elapsed time.

Figure 34.- Continued.



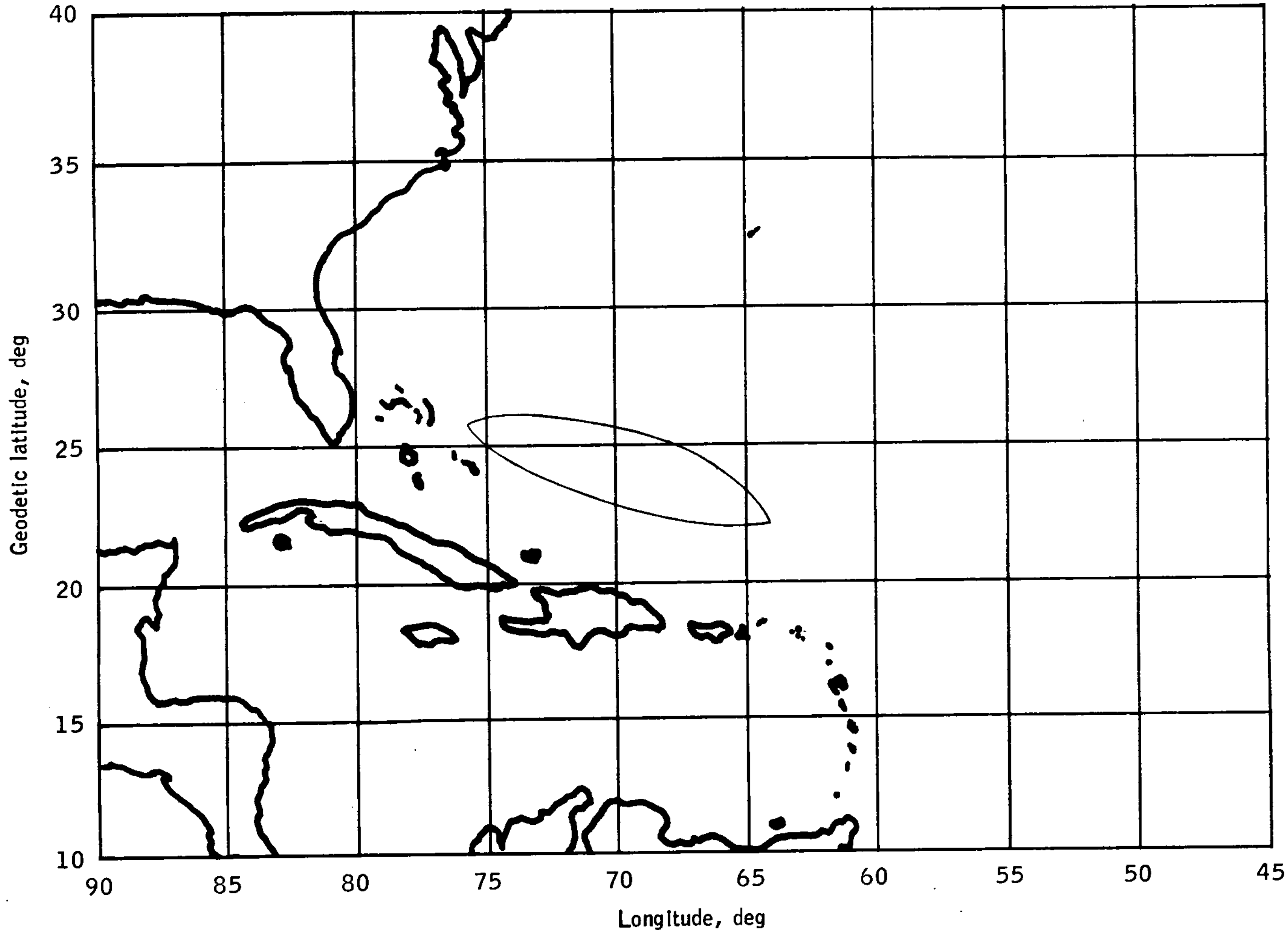
(d) Altitude versus ground elapsed time.

Figure 34.- Concluded.



(a) CM reentry groundtrack and maneuver envelope.

Figure 35.- Reentry trajectory, fifth period of activity.



(b) SM manuever envelope.

Figure 35.- Concluded.

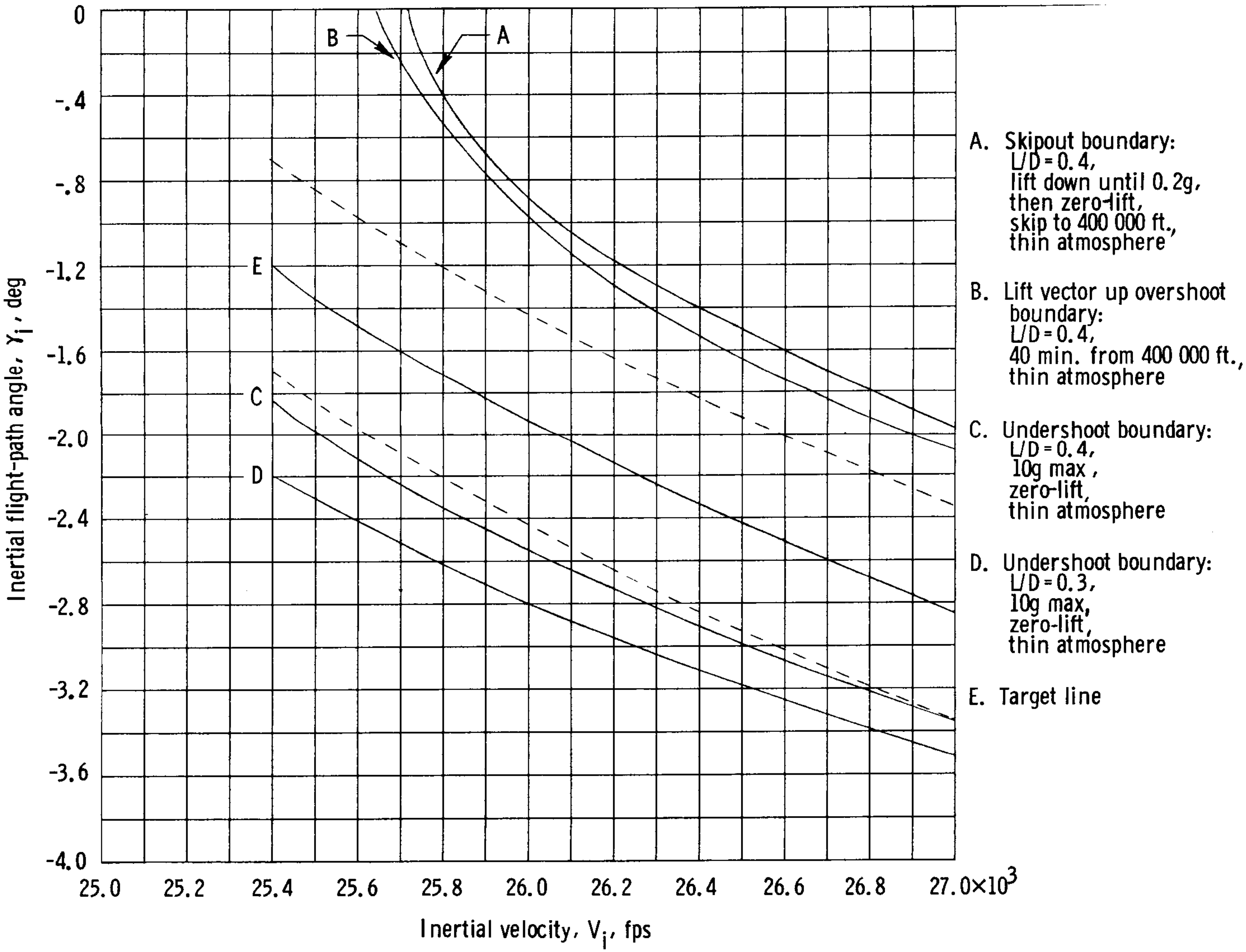


Figure 36. - AS-503A operational reentry corridor.

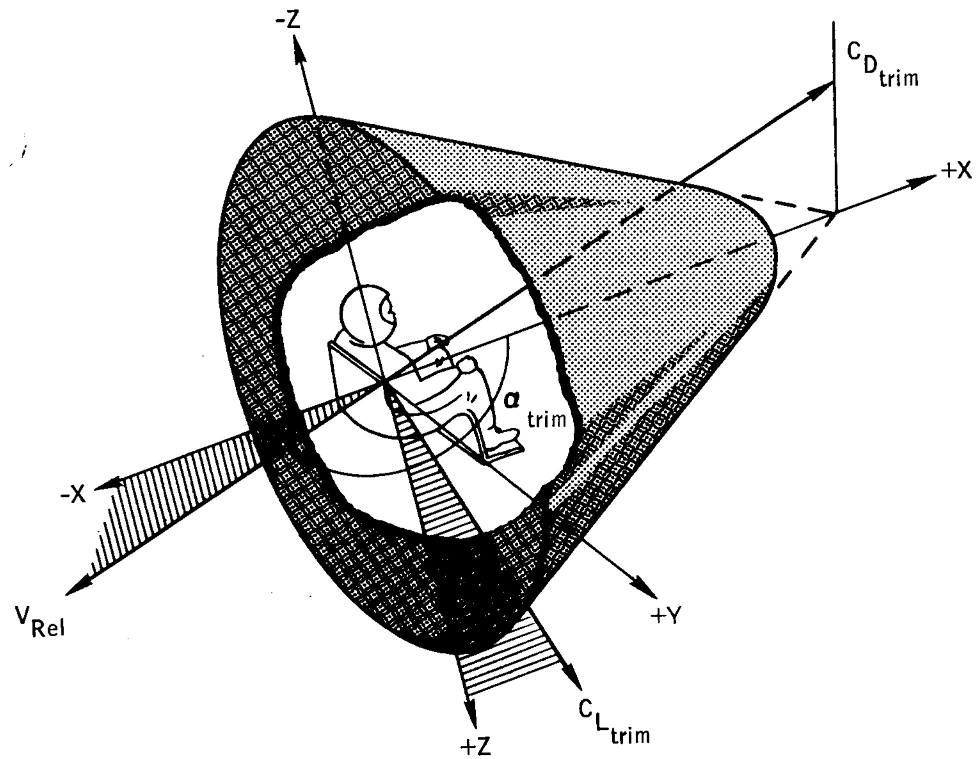


Figure 37.- CM aerodynamic force geometry.

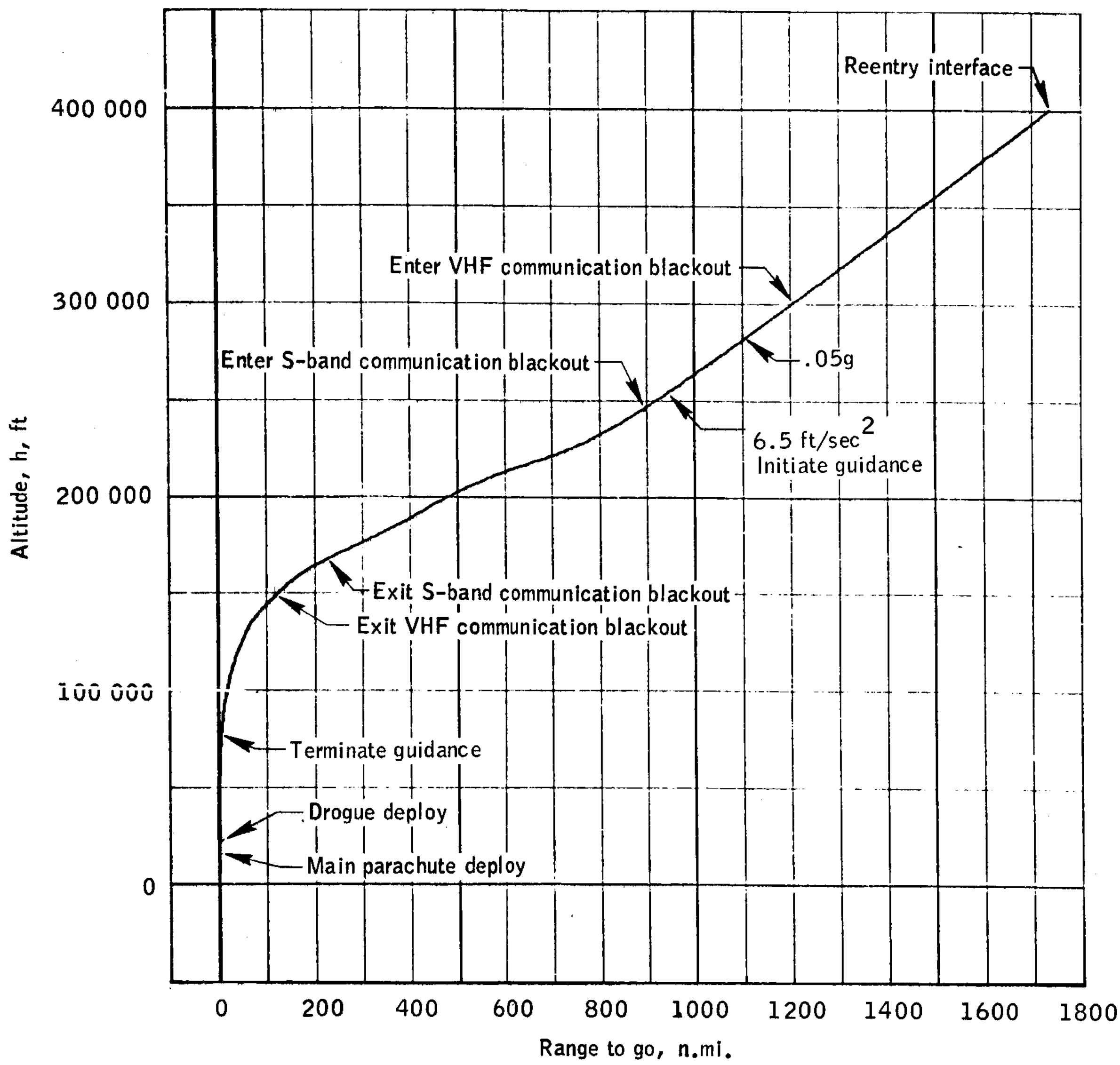
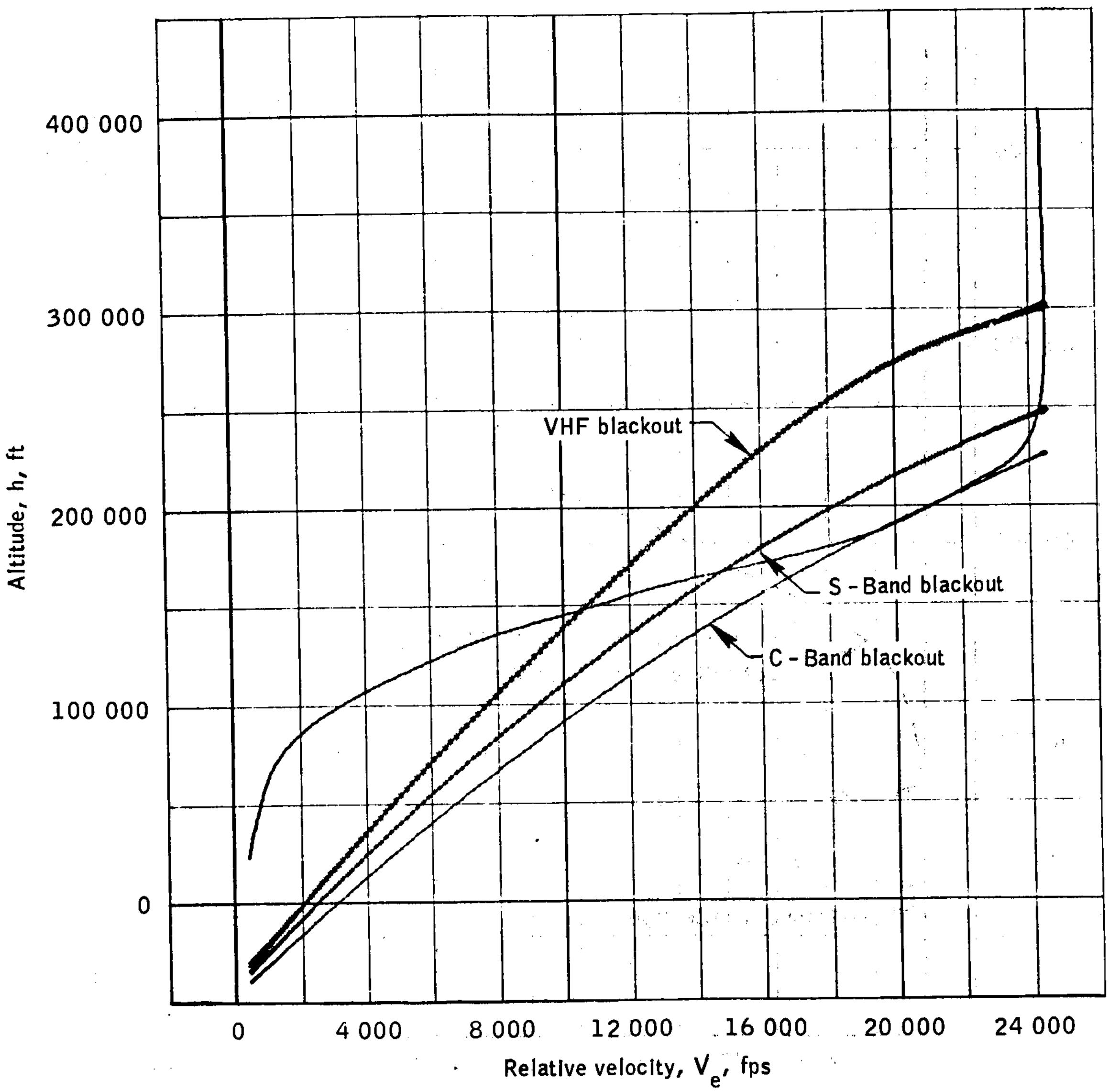
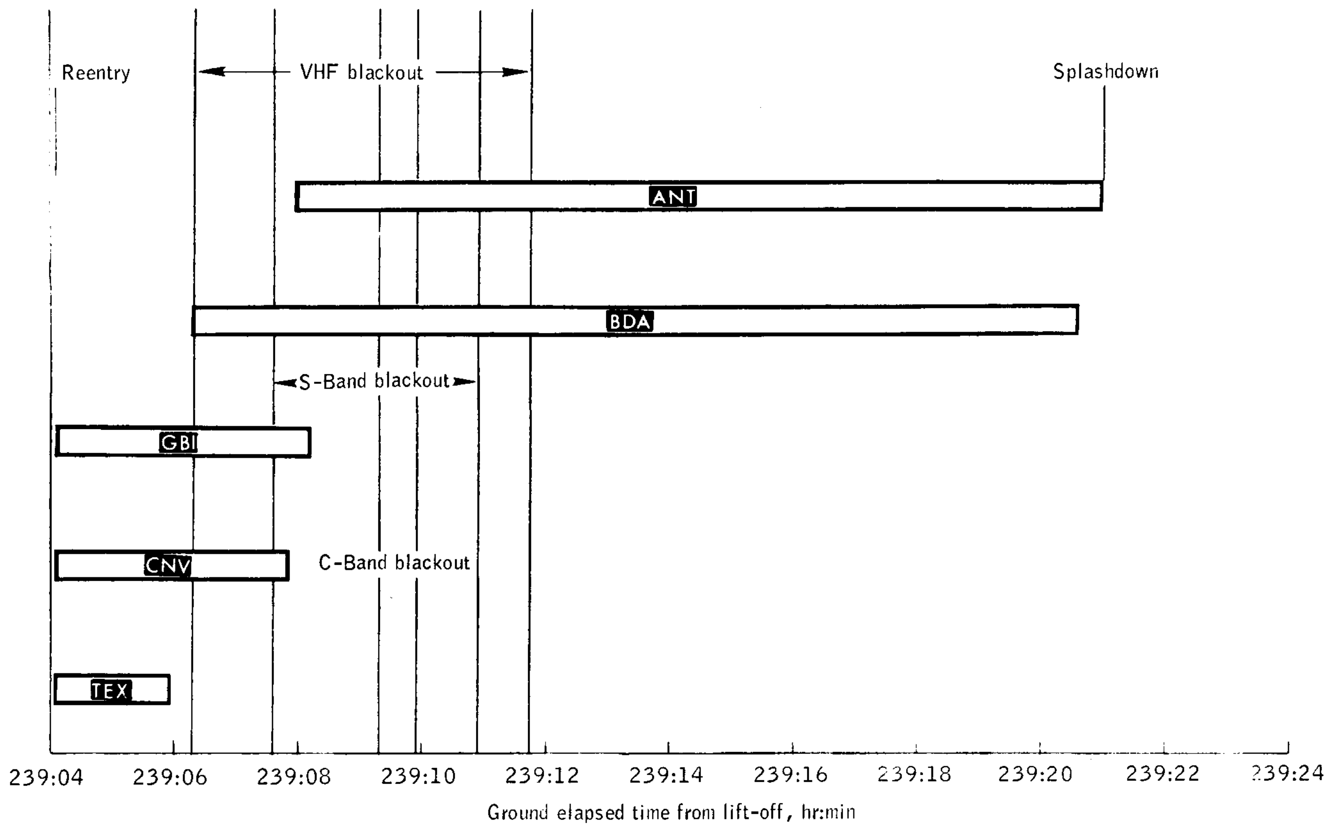


Figure 38.- Altitude - range profile.



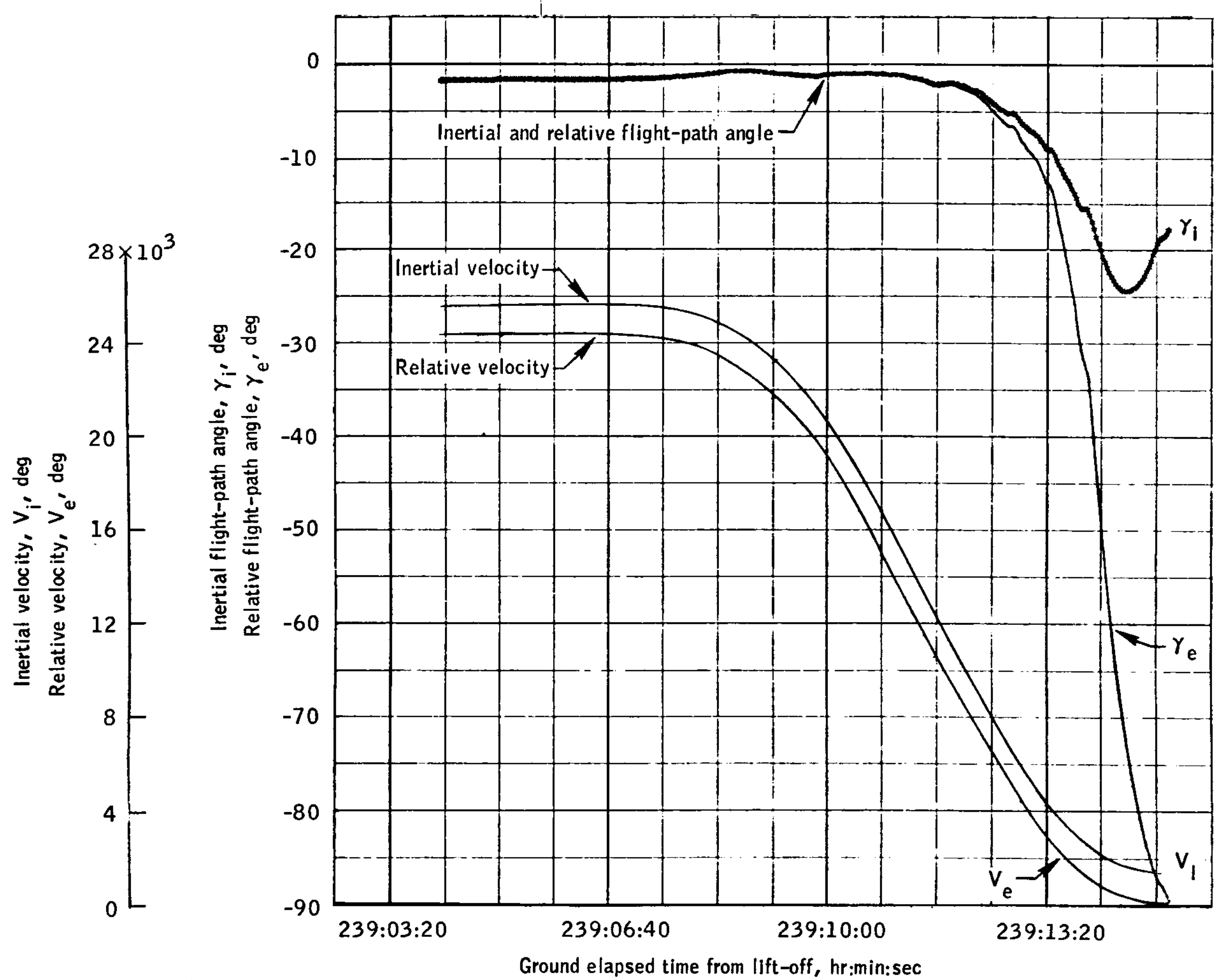
(a) Altitude versus relative velocity with communications blackout.

Figure 39.- Communications blackout for reentry.



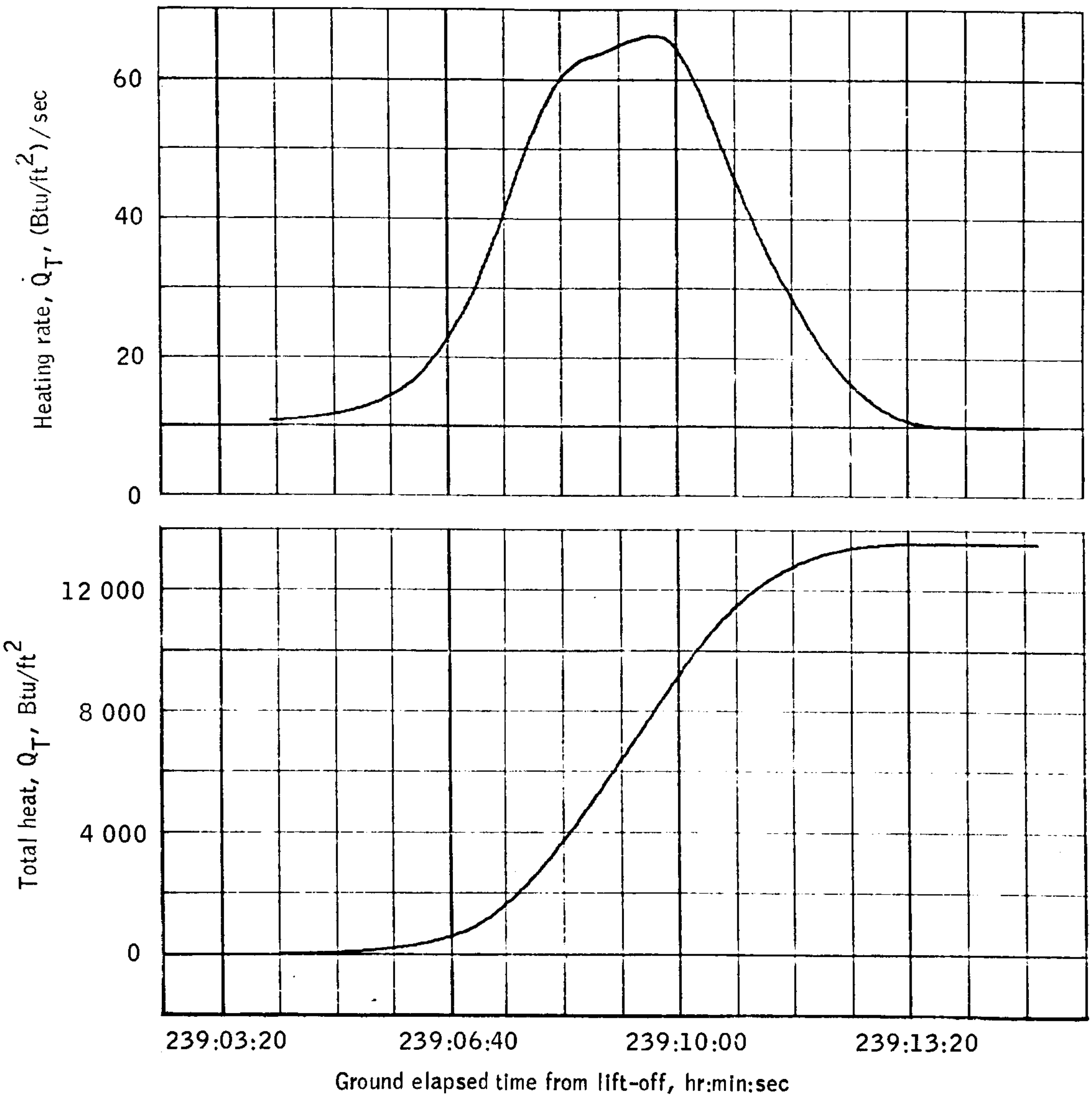
(b) Radar station availability for reentry tracking.

Figure 39.- Concluded.



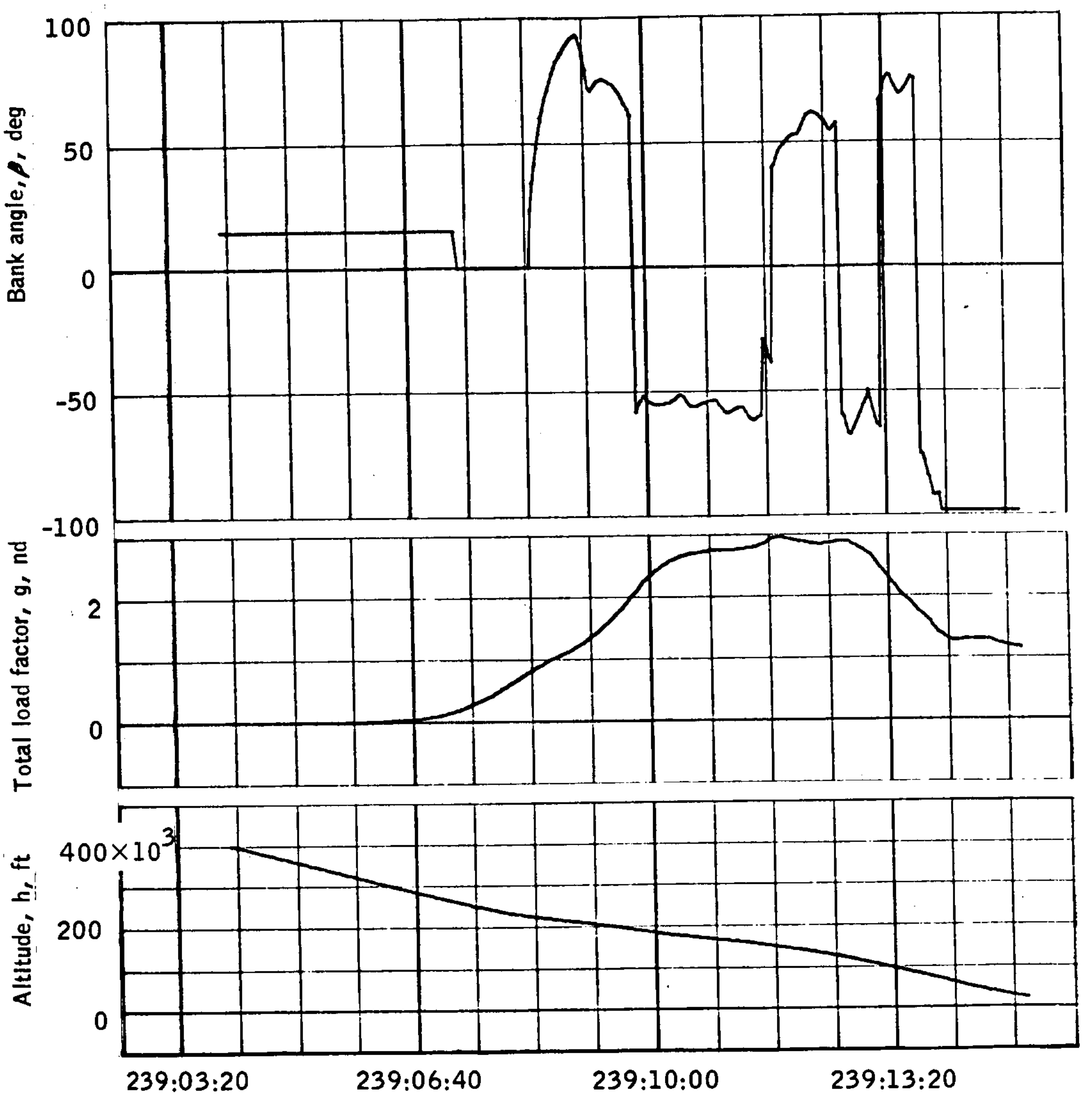
(a) Reentry velocity and flight-path angle time histories.

Figure 40.- Reentry phase for a SPS deorbit.



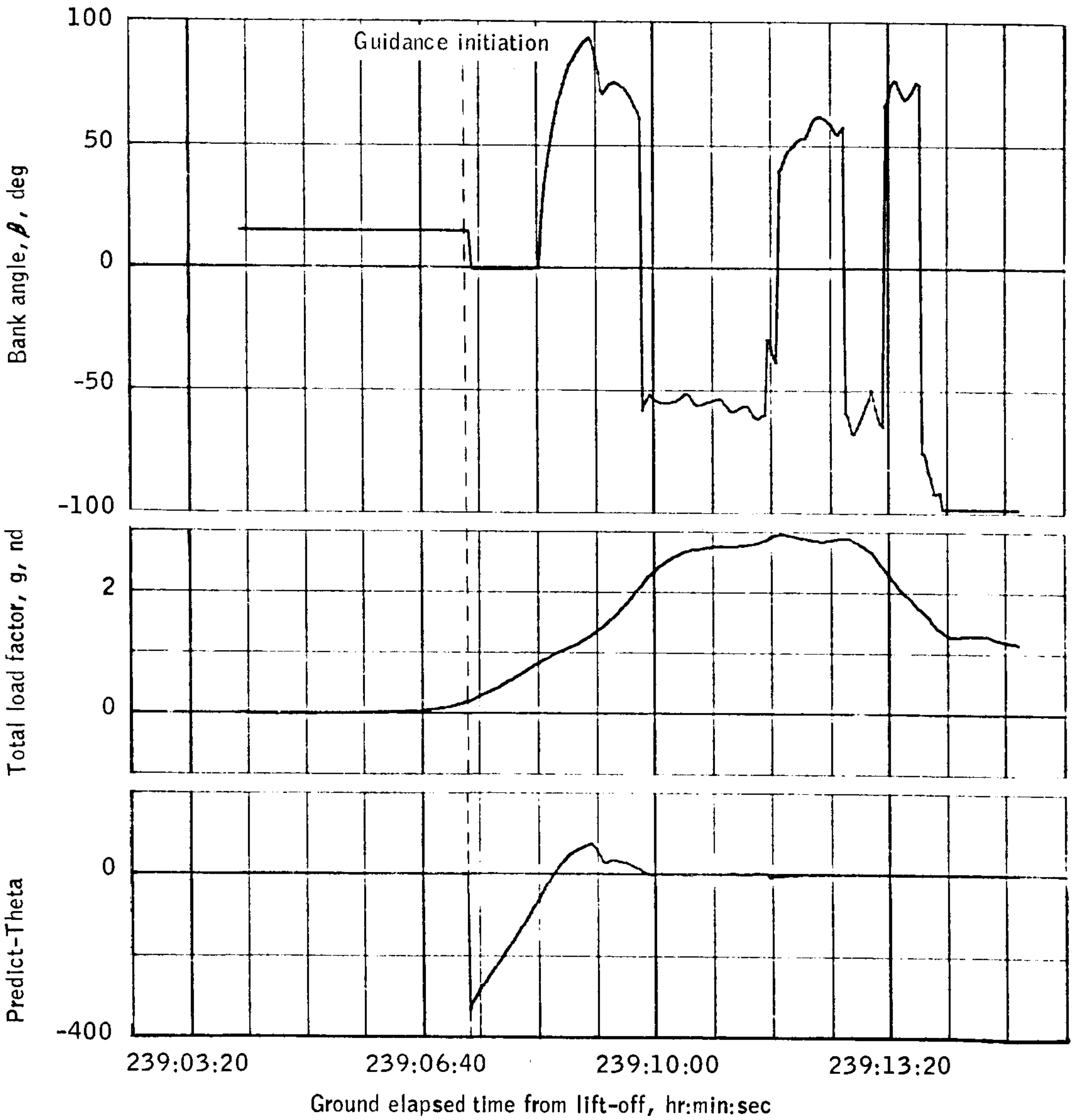
(b) Aerodynamic heating rate and heat load.

Figure 40.- Continued.



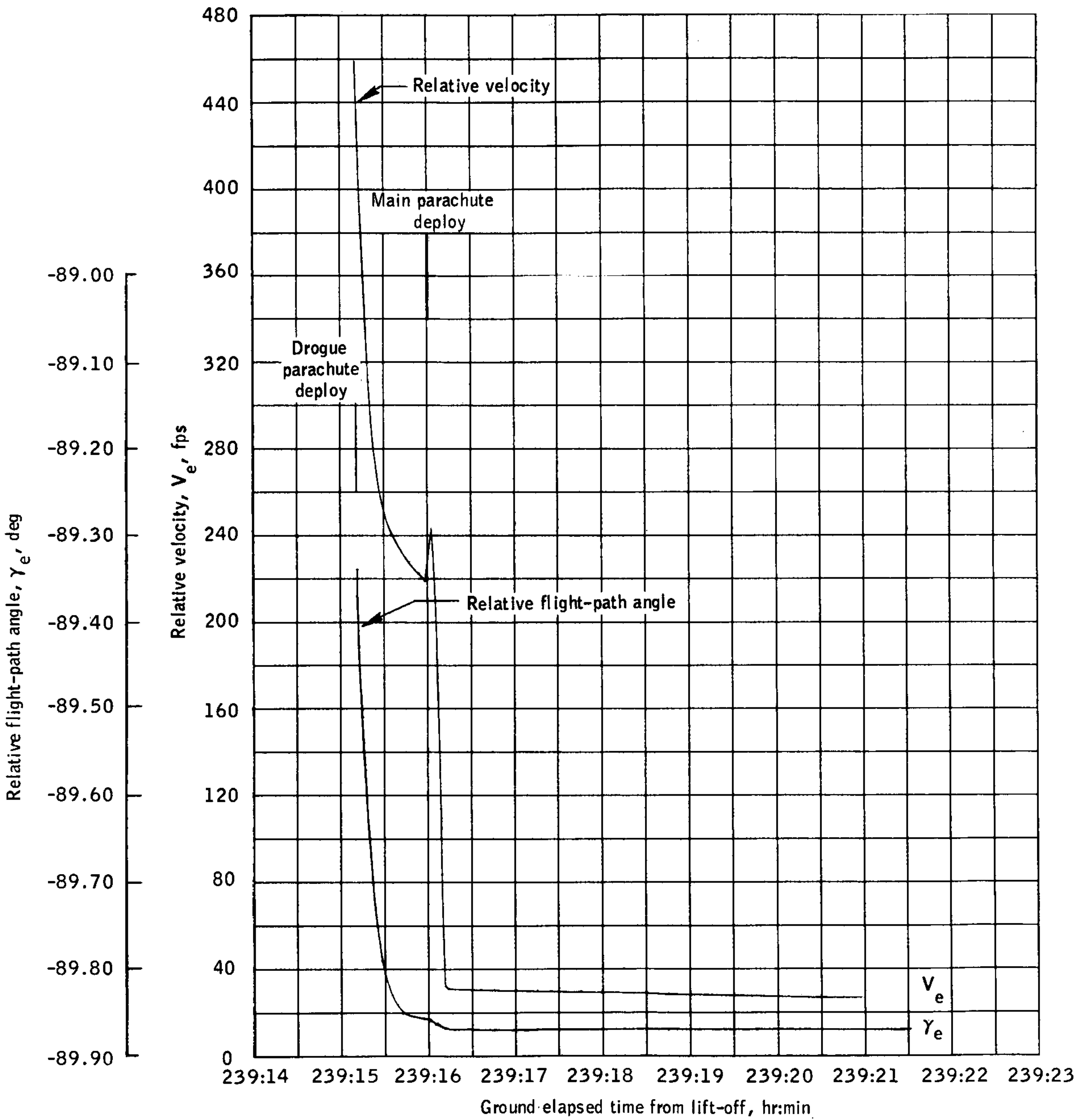
(c) Load factor, bank-angle, and altitude time histories

Figure 40.- Continued.



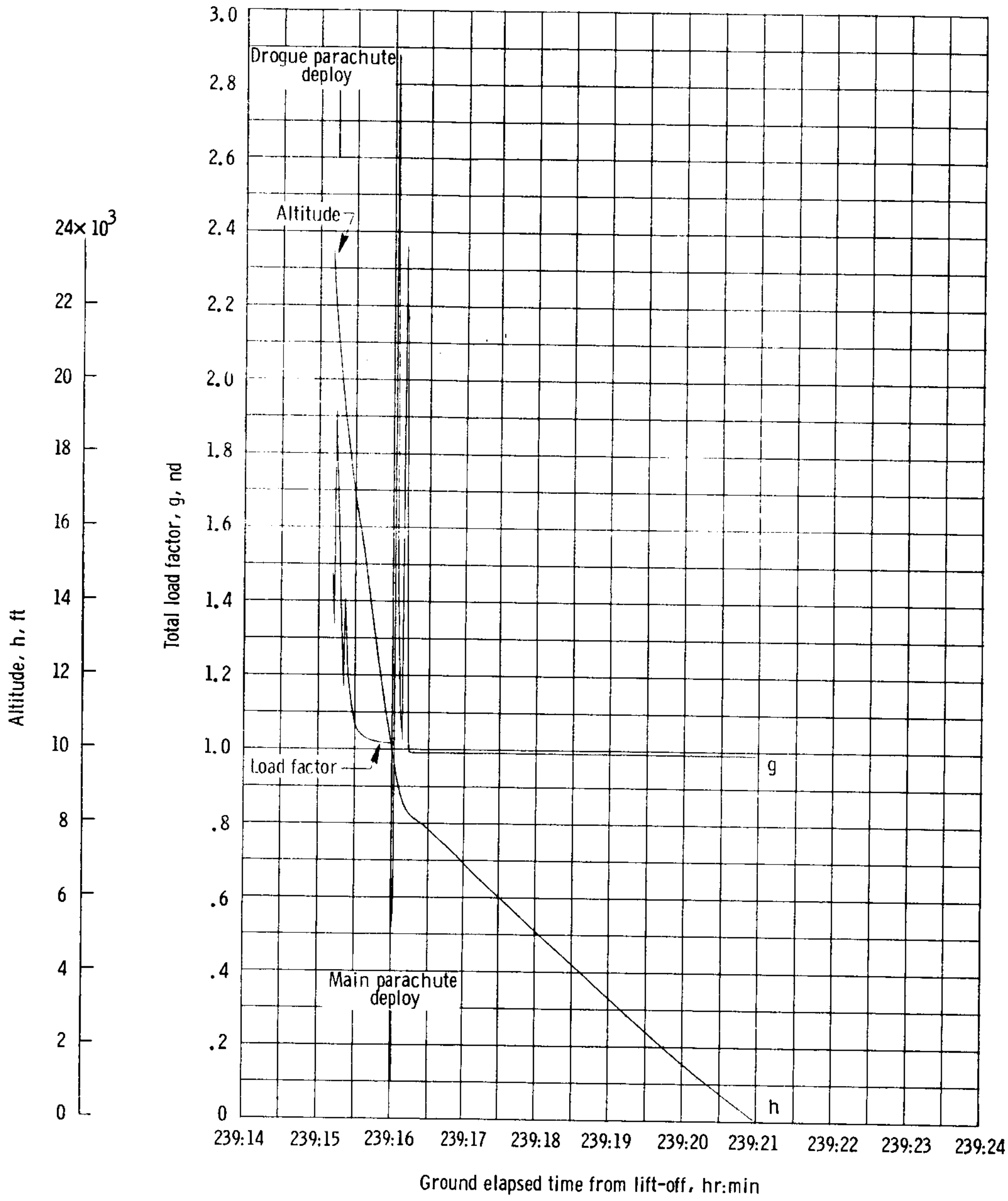
(d) Predicted range-range to go, load factor, and G and N commanded bank-angle time histories.

Figure 40.- Continued.



(e) Relative velocity and relative flight-path angle time histories for parachute phases.

Figure 40.- Continued.



(f) Altitude and load factor time histories for parachute phases.

Figure 40. - Concluded.

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